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HELICOPTER SECONDARY STRUCTURES RELIABILITY AND MAINTAINABILITY INVESTIGATION

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This report presents a highly responsive approach for dealing with reliability and maintainability (R&M) in the design stage of secondary structures for new helicopters. Specifically, the reader's attention is directed to the discussion of the use of the Failure Mode Effect and Criticality Analysis Technique - a step used in the probabilistic approach to R&M design - which appears to offer a readily usable improved design technique for secondary structures.

This report is one of two parallel efforts to improve the R&M of helicopter secondary structures. To further develop and verify quantitative design and test requirements, it is planned to integrate the results of these two parallel efforts with the results of a small hardware R&M investigation program. Advanced designs of selected secondary structures components will be developed and field tested.

The technical monitors for this contract were Major Andrew E. Gilewicz and Mr. Thomas E. Condon of the Military Operations Technology Division.

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Maintenance man-hours and downtime charged against secondary structures of aircraft have been excessive. It is assumed that this has been due to deficiencies in the design and test procedures for these components. This report investigates these problems and makes recommendations to alleviate them.

Standards and specifications were reviewed for application to secondary

20. ABSTRACT

structures on rotary-wing aircraft. A list was made up for detailed review.

Maintenance data were reviewed to select a "top ten" list of secondary structures for further analysis. A Failure Mode Effect and Reliability Analysis (FMERA) was performed using the maintenance data from the Sikorsky S-61 and projecting to similar items on the S-65. Results showed that: (1) secondary structures exhibit a reasonably constant failure rate with time, and (2) many of the S-65 failures could have been predicted by using the S-61 Work Unit Code data.

Three components among the "top ten" S-65 secondary structures were selected for test both as originally designed and in redesigned configurations to overcome field difficulties experienced.

It was found that field modes of damage can be reproduced in the test laboratory, provided that the reliability and maintainability reports are detailed enough to pinpoint the problem areas. For the most part, routine field data serve only to show that a problem exists for a particular component, but are scant as to the exact nature and location of failures. In order to design a test setup that will duplicate field failures, it is necessary to carry out a more detailed investigation of the exact nature and location of the failures.

This study indicates that design and test criteria for secondary structure components subject to handling use and abuse must be expanded to include the functional loads applied in handling. The usual flight and ground load criteria are inadequate for designing and testing adequate secondary structures of the hinged-opening variety.

PREFACE

The work for this study was authorized by Contract DAAJ02-72-C-0070 by the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, under the technical cognizance of Major Andrew E. Gilewicz and Mr. T. E. Condon.

The Sikorsky Aircraft personnel involved in performing or assisting in this study were:

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INTRODUCTION

Records show that 20 to 30 percent of non-depot maintenance man-hours on an Army helicopter are spent on repairing secondary structures. Secondary structures, such as panels, doors, floorings, fairings, cowlings, and maintenance platforms, are not flight-critical items. They do not carry aircraft structural loads. However, they are subject to aerodynamic, flight vibratory, acceleration, normal crew handling. maintenance, and abusive loads.

It was suspected, therefore, that the specifications for design of these components might be inadequate. In addition, if operational failure modes could be duplicated, test procedures for demonstrating the suitability of secondary structures could be improved.

This study was undertaken in order to:

(1) Evaluate the adequacy of existing design and test criteria as applicable to secondary structures.

(2) Evaluate the effectiveness of existing reliability and maintainability analytical techniques, such as use of existing R&M Data Bank and the performance of Failure Mode Effect and Reliability Analysis (FMERA) for minimizing secondary structures field problems in future aircraft.

DISCUSSION

REVIEW OF STANDARDS AND SPECIFICATIONS

Applicability

The applicability of fixed-wing and rotary-wing standards, specifications, and other documents (hereinafter referred to as documents) to the design of all helicopter secondary structures was determined progressively as follows:

1. The following index, list, and documents were searched:

   - Department of Defense Index of Specifications and Standards (1 July 1971)
   - List of Specifications and Standards (NAVAIR 00-25-544 of 1 July 1970)
   - ADS-1 Propulsion (Engine/Airframe) Interface Surveys
   - AMCP 706-134 Engineering Design Handbook Maintainability Guide for Design
   - AMCP 706-203 Engineering Design Handbook, Helicopters, Part III, Qualification Assurance
   - AR70-39 Criteria for Air Transport and Airdrop of Material
   - AR95-1 Army Aviation - General Provisions
   - DH2-3 Air Force Systems Command Design Handbook, Propulsion and Power

2. The above search resulted in the selection of the following for detail review for applicability to those secondary structure items that were potential candidates for further investigation:

SD-24H & J	General Specification for Design and Construction of Aircraft Weapon Systems - Volume II - Rotary Wing Aircraft
MIL-E-5272	Environmental Testing, Aeronautical and Associated Equipment
MIL-T-8679	Test Requirements, Ground Helicopter
MIL-S-8698	Structural Design Requirements, Helicopter
MIL-D-8706	Data and Tests, Engineering, Contract Requirements for Aircraft Weapon Systems

MIL-D-23222	Demonstration Requirements for Helicopters
MIL-I-83294	Installation Requirements, Aircraft Propulsion Systems, General Specification for
MIL-STD-210	Climatic Extremes for Military Equipment
MIL-STD-810	Environmental Test Methods
AMCP 706-203	Engineering Design Handbook, Helicopters, Part III, Qualification Assurance
DH 2-3	AFSC Design Handbook, Propulsion and Power

3. Detail review of the above documents resulted in selection of the following documents for research in depth for adequacy (or inadequacy) of design, test, and demonstration requirements for the candidate secondary structure items. Please note that some documents were added and others dropped as review progressed.

AMCP 706-202	Helicopter Engineering, Part 2, Detail Design (Note: listed for reference only since it has not been issued and is not available.)
AMCP 706-203	Engineering Design Handbook, Helicopters, Part 3, Qualification Assurance - December 1971.
SD-24H	General Specification for Design and Construction of Aircraft Weapon Systems - Volume II - Rotary Wing Aircraft - 13 March 1959.
SD-24J	General Specification for Design and Construction of Aircraft Weapon Systems - Volume II - Rotary Wing Aircraft - Change 1, dated 1 February 1966.
MIL-T-8679	Test Requirements, Ground, Helicopter.
MIL-S-8698(ASG)(-1)	Structural Design Requirements, Helicopters (Superseded by AR-56).
MIL-D-23222A(AS)	Demonstration Requirements for Helicopters.
MIL-I-83294(USAF)	Installation Requirements, Aircraft Propulsion Systems, General Specification for.
AR-56	NAVAIR Aeronautical Requirements; Structural Design Requirements (Helicopters) - 17 February 1970 (Supersedes MIL-S-8698).

Air Force Systems Command Design Handbooks, Series 1 and 2 as follows:

DH 1-2	General Design Factors (see note*).
DH 1-3	Personnel Subsystems (see note*).
DH 2-1	Airframe, First Edition, Rev. 5.
DH 2-2	Crew Stations and Passenger Accommodations, First Edition, Rev. 5.
DH 2-3	Propulsion and Power (see note*).

Note* DH 1-2, DH 1-3, and DH 2-3 were subsequently determined impertinent to this project.

Adequacy

A preliminary evaluation of the adequacy (or inadequacy) of design, test, and demonstration requirements was made through analysis of the selected documents, taking into consideration the types of malfunctions, deterioration and other unsatisfactory performance on record for secondary structures in general. The results of this specification analysis are reported in Appendix I with recommendations for improving the requirements. These recommendations were proven valid by subsequent test results as reported in the Evaluation section.

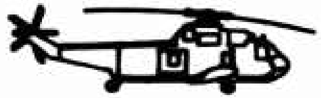
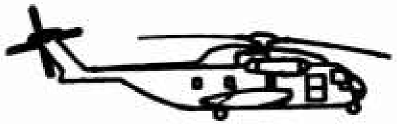
MAINTENANCE DATA ANALYSIS

A data search was conducted to determine the "top ten" items of secondary structure resulting in the highest number of maintenance man-hours to repair or replace, and the highest amount of aircraft downtime. The aircraft selected for the data search were the Army CH-54A, the Marine CH-53 A/D and the Air Force HH-53B/C. Certain basic data (gross weight, military designation, date of service introduction) of the several Sikorsky models involved in this study are presented in Table I.

The CH-54A data included 47,993 flight hours from 1 October 1967 to 1 October 1970, collected by Sikorsky Aircraft, under contract to the Army, in the Operational Reliability and Maintenance Engineering (ORME) program. They were the most complete data available, being based on 100% surveillance of the total aircraft population. The data search consisted of a detailed review of Discrepancy/Corrective Action Reports, which contain a detailed description of the part in question, maintenance man-hours and aircraft downtime required for repair or replacement, and a detailed description of the failure and corrective action. Maintenance man-hours and downtime were calculated for approximately 83 types of secondary structure. However, because of the minimum amount of secondary structure in the CH-54A, and because of its simplicity and accessibility, only one item from the CH-54A appears on the "top ten" list. Therefore, no further consideration was given to analyzing the CH-54A data.

The CH-53A/D data included 73,670 flight hours from August 1967 to March 1970 on the CH-53A, and 34,918 hours from November 1969 to March 1971 on

TABLE I. MODEL DESIGNATIONS

CONFIGURATION	GROSS WEIGHT (lb)	SIKORSKY DESIGNATION	MIL DESIGNATION (DATE ENTERED SERVICE)			
			AIR FORCE	ARMY	MARINE	NAVY
	19,000	S-61				SH-3A (1960)
	20,500					SH-3D (1966)
	19,100					SH-3G (1970)
	42,500	S-64		CH-54A (1967)		
	35,600	S-65			CH-53A (1964)	
	37,400		HH-53B (1967)			
	37,700		HH-53C (1968)			
	36,500				CH-53D (1969)	

the CH-53D. The data search consisted of a detailed review of the Maintenance Action Part Removal Details section of the Navy Maintenance and Material Management Report (3-M). This section contains: when discovered, how malfunctioned, corrective action, maintenance man-hours and elapsed maintenance time information. The 3-M data are consistent between CH-53A and CH-53D and appear to be complete. Maintenance man-hours and elapsed maintenance time were calculated for approximately 67 items of secondary structure.

The HH-53B/C data included 38,366 flight hours from January 1968 to June 1971. The data search consisted of a detailed review of the Maintenance Action How Malfunction Summary section of the Air Force Maintenance Management System (66-1) report. This contains: when discovered, how malfunctioned, action taken, and maintenance man-hours information. This report was considered to be the least reliable source of data because of the very large number of part removals listed for some Work Unit Codes (WUCs) and the complete lack of part removals reported for other WUCs.

The result of this study of the data bank of experience with secondary structures is given in Table II, showing the top ten secondary structure maintenance items.

In selecting the three secondary structures for test, windshields were eliminated from consideration because they were being investigated under a separate Army contract.

FAILURE MODE EFFECT AND CRITICALITY ANALYSIS

One of the primary purposes of this study was to determine whether the Failure Mode Effect and Criticality Analysis (FMECA) technique could be used to predict, during the design phase, the failures that were later experienced on the CH-53A/D. To simulate turning the clock back to the CH-53A/D design phase, the field failure mode and rate data on similar parts of the SH-3 series helicopters were used as inputs to the FMECA for the CH-53A/D. Every effort was made not to be prejudiced in this analysis by the known failure modes and rates of the CH-53A/D. Ten items were selected for evaluation to determine if the failure modes and failure rates of these items could be predicted during the design phase of the aircraft life cycle, rather than calculated after operational deployment. To accomplish this task, it was decided that the Failure Mode Effect and Criticality Analysis, with some modification, was the format most likely to result in a successful prediction.

The actual detailed FMECA is given in Appendix II. The step-by-step procedures used to carry out this analysis were as follows:

A reliability logic diagram was constructed for each of the ten items, showing the functional relationships of the basic components of an assembly. The reliability logic diagram lists all the parts to be included in an FMECA and generally progresses from the most basic part to a minor subassembly, if any, to the subassembly under investigation.

TABLE II. TOP TEN MAINTENANCE ITEMS

DRAWING NO.	NOMENCLATURE	CH-53A MH	CH-53A EMT	CH-53A FAIL-URES	CH-53D MH	CH-53D EMT	CH-53D FAIL-URES	H-53B/C MH	H-53B/C FAIL-URES	SUM OF CH-53A and D MH	SUM RANK	SUM EMT	SUM RANK	SUM FAIL-URES	SUM RANK
65205-09010-011	Hinge and Cover* M.R. Pylon	1361.0	662.9	39	287.0	190.0	15	-	-	1648.0	1	852.9	2	54	5
65207-03018-041	Personnel Door*	1168.8	729.3	32	146.8	118.3	21	-	-	1315.6	2	847.6	3	53	6
65207-09004-041/ -042/-044/-045	Work Platform* Assy	1153.1	966.6	112	110.7	90.9	39	256.0	26	1263.8	3	1057.5	1	151	1
65206-01009-105	Windshield Center	555.9	312.6	64	224.8	121.0	32	35.0	7	780.7	4	433.6	4	96	2
65206-01003-109	Windshield L.H.	432.0	230.8	38	208.6	106.9	30	2.5	3	640.6	5	337.7	5	68	3
65206-01003-110	Windshield R.H.	273.5	135.0	24	250.2	120.5	33	1.5	3	523.7	6	255.5	7	57	4
65205-09011-011	Cover and Slide Assy	249.1	133.8	18	192.0	126.1	13	106.7	9	441.1	7	259.9	6	31	8
65302-12503-041	EAPS Rear Frame	375.5	240.0	11	1.0	0.5	1	-	-	376.5	8	240.5	9	12	10
65207-08004-041/ -042	Cover Fuel Cell L.H. & R.H.	280.5	221.5	41	60.0	33.0	11	-	-	340.5	9	254.5	8	52	7
65207-02022-041	Door, Nose Gear Fwd Sect	275.5	206.5	21	5.0	4.0	1	-	-	280.5	10	210.5	10	22	9

MH - MAN-HOURS SPENT ON REPAIR/REPLACEMENT
EMT - ELAPSED MAINTENANCE TIME ON REPAIR/REPLACEMENT - HOURS

* Top 3 items for both maintenance man-hours and for elapsed maintenance time. These 3 items are recommended for test/qualification as originally designed and as designed to the new criteria/specifications.

Using the reliability logic diagram, the components of the subassembly to be investigated are listed and identified on the Failure Mode and Effect Analysis form, Appendix II. The identification section contains the following headings:

Column* (1) Name - The noun nomenclature as found in the illustrated parts breakdown.

(2) Identification No. - The number assigned to each component or subassembly on the reliability logic diagram.

(3) Drawing Reference Designation - The number assigned to each component or subassembly by the manufacturer.

(4) Reliability Logic Diagram Number - The number of the reliability logic diagram on which the component or subassembly appears.

*Note: These column numbers are used to facilitate the explanation of Table XXII and XXIII column headings. These column headings on subsequent tables of Appendix II are typical and, therefore, these column numbers are not repeated.

After the routine, but necessary, task of identifying the item to be analyzed, the qualitative portion of the FMEA is approached. The headings provide a logical development of the problems that may be anticipated with and design. They are:

Column* (5) Function - The function, intended or otherwise, that the component or subassembly performs.

(6) Failure Mode - A list of all failure modes anticipated for the subject component or subassembly. These failure modes are based on experience with previous parts of similar design, or they are based on the analyst's judgment when a completely new design or material is used.

(7) OperationPhase - Self-explanatory.

(8) Failure Effect on:

(a) Component/Functional Assembly - The effect of the selected failed component or subassembly.

(b) Next Higher Subsystem - The effect of the selected failure mode on the next higher subassembly.

(c) Uppermost System - The effect of the selected failure mode on the aircraft under consideration.

(9) Failure Detection Method - The manner in which the failure is most likely to be detected, such as inspection, warning

device, or adverse aircraft performance.

(10) Corrective Action Time Available/Time Required - The time between a component or subassembly failure and a catastrophic aircraft failure, and the time needed to recover or land following initial indication of a failure.

(11) Design Provisions to Reduce Criticality - Self-explanatory.

(12) Remarks - Any that may be helpful in pinpointing potential failures.

The criticality analysis is the quantitative portion of the FMECA and requires a data search for failure rates under operational conditions. The data search provides the most accurate information when a similar assembly can be found in the anticipated environment.

For this study, the criticality analysis was modified and called a reliability analysis. The identification section is identical with the first four items of the FMEA.

The headings in the reliability analysis are as follows:

Column* (13) Function - The same as item (5) of the FMEA.

(14) Failure Mode - The same as item (6) of the FMEA.

(15) Operational Phase - The same as item (7) of the FMEA.

(16) Failure Effects - The failure effects are understood to be the same as found in the FMEA and are severe enough to require repair and replacement of the subassembly under investigation.

(17) Reliability Data Source Code - Identification of the reports used to determine failure mode ratio and generic failure rate.

(18) Probability of Failure Effects - Deleted for this study. Probability is 1.00 since we are dealing with failure.

(19) Failure Mode Ratio - The percentage that each failure mode contributes to the total failure rate.

(20) Environment Ratio - The factor which adjusts the generic failure rate for differences between environmental stresses when the generic failure rate was measured and environmental stresses under which the component is going to be used.

(21) Operational Ratio - The factor which adjusts the generic failure rate for differences between operational stresses when the generic failure rate was measured and operational

stresses under which the component is going to be used.

(22) Generic Failure Rate, Failures/One Hour - The failure rate per flight hour of very similar or identical subassemblies installed on operational aircraft. The total generic failure rate per hour is repeated for each failure mode under item (13).

(23) Operating Time, Hours or Cycles - Deleted for this study. All calculations are on a per-one-hour basis.

(24) Failure Mode Contribution - The failure rate that can be expected from each failure mode. It is ($\alpha K_E K_A \lambda_G$).

(25) Component Criticality Number, C_r - The total repair and replacement rate predicted for the subject subsystem; it is equal to $\Sigma(\alpha K_E K_A \lambda_G)$.

(26) (Under Column (18)) - Hazard Level - The hazard resulting from a component or subassembly failure is based on the definition in paragraph 3.14 of MIL-STD-882: a qualitative measure of hazard level stated in relative terms.

(a) Category I - Negligible
...will not result in personnel injury or system damage.

(b) Category II - Marginal
...can be counteracted or controlled without injury to personnel or major system damage.

(c) Category III - Critical
...will cause personnel injury or major system damage, or will require immediate corrective action for personnel or system survival.

(d) Category IV - Catastrophic
...will cause death or severe injury to personnel, or system loss.

It should be noted that under the failure mode ratio heading, only the subsystem is given a quantitative breakdown of failure. This is because the available data on components and subassemblies do not include detailed information in large enough quantities to be reliable. Therefore, the failure mode ratio for components or subassemblies (below the double line on the form) is presented qualitatively according to Table III.

TABLE III. FAILURE MODE RATIO DEFINITIONS	
Actual	100%
Very Probable	60% to 100%
Probable	10% to 60%
Possible	3% to 10%
Not Very Possible	0% to 3%
None	0%

The results of the analysis in terms of failure rates are given in column 22 of the Appendix II.

The FMECA as modified, during the initial phase of this study, to the FMERA, Failure Mode Effect and Reliability Analysis, has presented a logical and easy to understand development of the design weaknesses and potential failures of an item of secondary structure. Failure rates based on data from the S-61 helicopter provided a good basis for predicting failure rates on the S-65. These predicted failure rates were then ranked as were the actual S-65 failure rates, and a comparison was made.

The comparative data as presented in Table IV shows all the rankings to be within three numbers of each other with the single exception of the first item (Housing Assembly - Ranking Difference = 5 Numbers).

TABLE IV. S-65 FAILURE RATE RANKING				
	Predicted From FMERA Using Earlier S-61 Field Data		Actual From S-65 Field Data	
	Rate	Rank	Rate	Rank
Housing Assy	.0040	6	.0148	1
Hinge & Cover	.0050	5	.0126	2
Slide & Cover	.0065	2	.0112	4
Nose Gear Door	.0011	9	.0027	7
Personnel Door, Lower	.0186	1	.0114	3
Fuel Cell Cover	.0017	7	.0000	10
Service Plat., Sponson	.0054	3	.0081	6
Work Platform, M.R.P.	.0053	4	.0085	5
EAPS Rear Frame	.0015	8	.0005	8
Compass Support	.0004	10	-	9

It should be pointed out that the "failure rate" as presented throughout this study is not the classical one because the time to failure of individual parts is not available from the base data. In this study, failure rate is defined as the total fleet aircraft time divided by the total number of failures reported.

A statistical comparison was conducted of the Table IV data to test the validity of using the FMERA for predicting secondary structure reliability/maintainability. The fundamental notion is that if there is insufficient evidence to reject the hypothesis that the predictions are correct, we will accept it. The measure used to determine whether or not there is enough evidence for rejection is the generalized likelihood ratio, $\underline{\lambda}$.

Let $X_1, X_2, \ldots, X_n$ be a random sample of size n from a density $f(X, \theta_1, \theta_2, \ldots, \theta_k)$ that satisfies quite general regularity conditions, and suppose Ω is k-dimensional. Suppose that it is desired to test the hypothesis

$$H_o: \ \theta_1 = \theta_1^o, \ \theta_2 = \theta_2^o, \ \ldots, \ \theta_t = \theta_t^o \quad t<k$$

where $\theta_1^o, \theta_2^o, \ldots, \theta_t^o$ are known numbers.

When H_o is true, $-2 \log \underline{\lambda}$ is approximately distributed as chi-square, $\chi^2_{\alpha,t}$, with t degrees of freedom when n is large.[1, 2, 3] Ω in the above theorem is the entire parameter space and $\underline{\lambda}$ is the generalized likelihood ratio.

The generalized likelihood-ratio is the quotient

$$\lambda = \frac{L(\hat{\omega})}{L(\hat{\Omega})} \tag{1}$$

where $L(\hat{\omega})$ is the maximum of the likelihood function in the region ω with respect to the parameters (the region for which the hypothesis under test is true) and $L(\hat{\Omega})$ is the maximum of the likelihood function in the region Ω with respect to the parameters.

Assuming a constant failure rate, the Poisson density distribution applies:

$$f(X_i, \lambda_i) = \frac{e^{-\lambda_i T}(\lambda_i T)^{X_i}}{X_i\,!} \tag{2}$$

The likelihood function for this density is $L(\Omega) = \pi_i f(X_i, \lambda_i)$

$$= e^{-\Sigma_i \lambda_i T} \pi_i \left[\frac{(\lambda_i T)^{X_i}}{X_i\,!} \right] \tag{3}$$

where λ_i is the failure rate of the i th component, X_i is the observed number of failures of the i th component, and T is the accumulated time in which X_i failures were observed.

The maximum value, $L(\hat{\Omega})$ is,

$$L(\hat{\Omega}) = e^{-\Sigma_i X_i} \pi_i \left[\frac{(X_i)^{X_i}}{X_i\,!} \right] \tag{4}$$

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[1] Mood, A. M., & Graybill, F. A., INTRODUCTION TO THE THEORY OF STATISTICS, Second Edition, New York, McGraw-Hill, 1963, p. 301.

[2] All logs are to the base e.

[3] $P(-2 \log \underline{\lambda} > \chi^2_{\alpha,t}) = \alpha$, where α is the probability of rejecting the hypothesis when it is true.

To find $L(\hat{\omega})$, λ_i is set equal to λ_i^{o}.

Since L is single valued in the subspace ω

$$L(\hat{\omega}) = e^{-\sum_i \lambda_i} \prod_i \left[\frac{(\lambda^{o}_i T)^{X_i}}{X_i !} \right] \tag{5}$$

The generalized likelihood ratio, $\underline{\lambda}$, is given by

$$\underline{\lambda} = e^{-\sum_i (\lambda_i^{o} T - X_i)} \prod_i \left[\frac{\lambda_i^{o} T}{X_i} \right]^{X_i} \tag{6}$$

To make use of the theorem stated above, the following condition must be satisfied for the hypothesis under test to be accepted:

$$-2 \text{ Log } \underline{\lambda} < \chi^2_{\alpha,t} \tag{7}$$

Substituting (6) into (7) results in

$$-2 \text{ Log } \underline{\lambda} = -2 \left[-\sum_i (\lambda_i^{o} T - X_i) + \sum_i X_i \text{ Log } \frac{\lambda_i^{o} T}{X_i} \right]$$

$$= 2 \sum_i (\lambda_i^{o} T - X_i) + 2 \sum X_i \text{ Log } \frac{X_i}{\lambda_i^{o} T} \tag{8}$$

Thus, we accept the hypothesis under test with the type I error, α, if

$$2 \sum_i (\lambda_i^{o} T - X_i) + 2 \sum_i X_i \text{ Log } \frac{X_i}{\lambda_i^{o} T} < \chi^2_{\alpha,t} \tag{9}$$

Thus, for the data shown in Table IV and the hypothesis that the predictions shown there are correct, the following condition must be satisfied: $-2 \text{ Log } \underline{\lambda} < \chi^2_{\alpha,t}$. Using equation (8) to calculate $-2 \text{ Log } \underline{\lambda}$ and the fact that we accept the hypothesis with type I error if equation (9) is satisfied, the following calculations were made. From Table V, the nine items listed constitute the sample size, t = 9, and from equation (9) the calculated

value of -2 Log $\underline{\lambda}$ is 1375. Since for α = .005 and t = 9, $\chi^2_{.005,9} = 23.589$, it is easily seen that -2 Log $\lambda >> \chi^2_{.005,9}$ and that for t = 9, α is much less than .005.[4] As a result, it can be seen from the preceding calculation that an α much less than .005 is needed. That is, based on observed failure rates, the predicted failure rates cannot be proven wrong. The observed failure rate can be reasonably forecast with the predicted component failure rates.

While recognizing that the sample size used for this test was small in conventional terms, it is felt that the statistical method and the resulting conclusions are valid.

TABLE V. S-65 FAILURE RATES - FMERA AND FIELD EXPERIENCE COMPARISON

Name	Predicted from FMERA Using Earlier S-61 Field Data	Rank	Actual S-65[1] Field Data	Rank
Housing Assembly	.0040	6	.0114	7
Hinge & Cover	.0050	5	.0394	2
Slide & Cover	.0065	2	.0408	1
Nose Gear Door	.0011	9	.0089	8
Personnel Door - Lower	.0186	1	.0375	3
Fuel Cell Cover	.0017	7	.0148	5
Service Platform - Sponson	.0054	3	.0179	4
Work Platform - M.R.P.	.0053	4	.0132	6
EAPS Rear Frame	.0015	8	.0031	9

(1) Failure rates given in this table are the average instantaneous values associated with the last three quarters of 1972. Instantaneous rather than cumulative values given in Table IV were used for the test because they more closely represent what the hardware is doing now. The compass support listed in Table IV was deleted from the above analysis because of insufficient U.S. Navy S-65 data. It will be noted that rankings are within 3 numbers without exception.

[4] Consult Table H-3b of AMCP 702-3, Quality Assurance Reliability Handbook 1968 for values of $\chi^2_{\alpha,t}$. In Table H-3b, ν represents the degrees of freedom. Hence t is synonymous with ν.

CONSTANT FAILURE RATE DISCUSSION

As an integral part of this study, it was determined that items of secondary structure exhibit a reasonably constant failure rate with time, where failure rate is defined as total aircraft fleet time divided by total number of failures (times to failure of individual parts were not available). To test this statement, actual failure rates for 3 Navy models (SH-3A, SH-3D, SH-3G) and another statistical test of hypothesis was used. This was done by making the assumption that the hypothesis is true (the failure rate is constant regardless of whether the aircraft is an SH-3A, SH-3D, or SH-3G) and calculating the type I error which results. Type I error is the error which results from rejecting the proposed hypothesis when it should have been accepted. Table VI presents the observed failure rate per flight hour for the three models of SH-3's, and shows the failure rate for each Work Unit Code as essentially constant regardless of the fact that SH-3A aircraft have on the average higher accumulated hours than the SH-3D's and SH-3D's have more average accumulated hours than SH-3G's. As a result, there exists a particular value for component's true failure rate regardless of whether the aircraft is an SH-3A, SH-3D, or SH-3G.

TABLE VI. THE OBSERVED FAILURE RATES PER FLIGHT HOUR FOR SH-3A/D/G

Model	SH-3A	SH-3D	SH-3G
Hours	19,340	19,197	13,345
Work Unit Code[1]			
	Personnel Door		
1122A	.0039	-	.0058
1122F	.0028	-	.0027
11227	.0017	.0140	.0150
11228	.0059	.0069	.0084
Complete Assembly	.0143	.0209	.0319
	Transmission Service Platform		
1123D	.0060	.0076	.0073
1123E	.0033	.0030	.0043
1123Q	.0080	.0063	.0087
1123S	.0050	.0043	.0057
1123T	.0022	.0020	.0030
11236	.0037	.0043	.0099
11238	-	-	.0018
Complete Assembly	.0282	.0275	.0407

[1] The Work Unit Codes are subassemblies of the Personnel Door and the Transmission Service Platform respectively.

The first step is to verify that the components we are testing are the same on each aircraft. Having verified this, we can then make the assumption that the observed failure rates are identically distributed and make use of the central limit theorem. As a result, a generalized likelihood ratio test will be used to test the hypothesis

$$H_o: \mu = \mu_o \quad 0 < \sigma < \infty$$

where μ_o is a given number assuming that we have a sample of n observations, $X_1, \ldots, X_n$, from a normal population.

The parameter space Ω is the half plane $\Omega = \{\mu, \sigma^2: -\infty < \mu < \infty;\ 0 < \sigma^2 < \infty\}$.

The subspace ω characterized by the null hypothesis is the vertical line

$\mu = \mu_o$, i.e.,

$$\omega = \{\mu, \sigma^2: \quad \mu = \mu_o;\ 0 < \sigma^2 < \infty\}$$

where μ_o is a given number.

We shall test H_o by means of the generalized likelihood-ratio. The likelihood is

$$L = \left(\frac{1}{\sqrt{2\pi\ \sigma^2}}\right)^n e^{-\frac{1}{2}\sum_i (X_i - \mu)/\sigma^2} \tag{10}$$

It can be shown that the values of μ and σ^2 which maximize L in Ω are

$$\hat{\mu} = \frac{1}{n} \sum_i X_i = \overline{X}$$

$$\hat{\sigma}^2 = \frac{1}{n} \sum_i (X_i - \overline{X})^2 \tag{11}$$

Substituting these values in L, we have

$$L(\hat{\Omega}) = \left(\frac{1}{(2\pi/n) \sum_i (X_i - \overline{X})^2}\right)^{n/2} e^{-(n/2)} \tag{12}$$

To maximize the L in ω, we put $\mu = \mu_o$, and the only remaining parameter is σ^2; the value of σ^2 which then maximizes L is readily found to be

$$\hat{\sigma}^2 = \frac{1}{n} \sum_i (X_i - \mu_o)^2$$

which gives

$$L(\hat{\omega}) = \left(\frac{1}{(2\pi/n) \sum_i (X_i - \mu_o)^2} \right)^{n/2} e^{-(n/2)} \tag{13}$$

The ratio of (13) to (12) is the generalized likelihood-ratio:

$$\underline{\lambda} = \left[\frac{\sum_i (X_i - \overline{X})^2}{\sum_i (X_i - \mu_o)^2} \right]^{n/2} \tag{14}$$

The next step is to obtain the distribution of $\underline{\lambda}$, use that distribution to determine a number A so that the critical region $o < \lambda < A$ will give the probability α. Table VII summarizes the results of this step and shows the interval over which μ_o can vary as a function of α, the type I error. As a result, Table VII shows the range of true failure rates which produce the scatter of observed failure rates of Table VI and the probability of error, α, associated with accepting the hypothesis that the true failure rate is in the indicated range. The greater the range of true failure rates, the more accurately they encompass the scatter of observed failure rates, and the smaller the chance of error (α) in assuming the true failure rate is within that range.

What we are saying is that the true failure rate that produces the scatter of observed failure rates in Table VI is not unique. In fact, any number of "constant" failure rates could produce the observed scatter. By "constant," we mean it is possible to associate a single value for the failure rate with the population of observed values. Since we have a small sample, the sample could exist in many different theoretical populations. We therefore have a range of population parameters which we call true failure rates provided in Table VII that could produce the observed scatter of Table VI. The range of true failure rates depends on the error you incur by making such associations.

For example, when $\alpha = 0.5$, the failure rate on the personnel door is .02237±.00416(19%) and the service platform is .03213±.0035(11%). There is only a 50% chance (based on SH-3A, 3D and 3G data) that the assumption of a single failure rate in these ranges will be in error. On this basis, the assumption of equal failure rates for the three models, and, therefore, a failure rate independent of time is considered to be a reasonable assumption.

TABLE VII. RANGE OF PERMISSIBLE VALUES OF FAILURE RATE, μ_o

Work Unit Code	α = .10	α = .20	α = .40	α = .50	α = .60
			Personnel Door		
1122A	(.0000,.0082)*	(.0000,.0065	(.0014,.0050)	(.0018,.0046)	(.0022,.0043)
1122F	(.0000,.0045)*	(.0001,.0036)	(.0009,.0028)	(.0011,.0026)	(.0013,.0024)
11227	(.0000,.0105)*	(.0025,.0186)	(.0060,.0151)	(.0071,.0140)	(.0079,.0132
11228	(.0049,.0092)	(.0057,.0084)	(.0063,.0078)	(.0182,.0265)	(.0066,.0075)
Complete Assembly	(.0075,.0373)	(.0128,.0320)	(.0170,.0278)	(.0182,.0265)	(.0192,.0255)
			Transmission Service Platform		
1123D	(.0055,.0084)	(.0060,.0079)	(.0064,.0075)	(.0066,.0074)	(.0067,.0073)
1123E	(.0024,.0047)	(.0028,.0040)	(.0031,.0035)	(.0032,.0038)	(.0033,.0038)
1123Q	(.0056,.0097)	(.0063,.0090)	(.0069,.0084)	(.0071,.0082)	(.0072,.0081)
1123S	(.0038,.0062)	(.0043,.0058)	(.0046,.0054)	(.0047,.0053)	(.0048,.0052)
1123T	(.0015,.0033)	(.0018,.0030)	(.0021,.0027)	(.0022,.0027)	(.0022,.0026)
11236	(.0002,.0117)	(.0022,.0097)	(.0039,.0081)	(.0044,.0076)	(.0048,.0072)
11238	(.0000,.0024)*	(.0000,.0017)*	(.0000,.0012)	(.0001,.0011)	(.0002,.0010)
Complete Assembly	(.0196,.0447)	(.0240,.0402)	(.0276,.0367)	(.0286,.0356)	(.0295,.0348)

*Those open intervals with negative lower bounds were set to .0000 even though negative values for the lower bound were permitted.

As part of the Reliability Analysis portion of the Failure Mode Effect and Reliability Analysis, hazard levels were assigned to all failure modes of components and subassemblies according to the definition of paragraph 3.14 of MIL-STD-882. From those definitions, it was determined that failures of items of secondary structure occurring on the ground could be classified as Category I - negligible, and that failures of items of secondary structure occurring in flight could be classified as Category II - marginal. There were no failures investigated during this study that had a higher hazard level than Category II.

Since items of secondary structure exhibit a reasonably "constant" failure rate and because they do not exhibit a hazard level higher than Category II, they should remain "on condition" replacement parts.

Following completion of the Failure Mode Effect and Reliability Analysis for the "top ten" items, three of these items were selected for redesign using the information available in the FMERA and field experience analysis.

The three items of secondary structure selected for test were:

Hinge and Cover Assembly, Part No. 65207-09010-011
Lower Personnel Door, Part No. 65207-03018-041
Work Platform Assembly, Part No. 65207-09004-041

EVALUATION

Reliability and Maintainability Analytical Techniques

The reliability and maintainability techniques used in this investigation were: (1) the use of the data bank of experience to determine areas where significant product improvements can be achieved through additional engineering effort, (2) the use of the failure mode and effect analysis as a means of predetermining areas requiring engineering attention, and (3) trade-off analysis.

The data bank of experience was found to be useful for ranking secondary structures in terms of failures per thousand hours of flight, maintenance man-hours per flight hour, and elapsed hours per maintenance action. The field data generated by the military data collection systems are deficient, however, in terms of defining the modes of failure, their exact locations, and the possible causes of the failures. To determine these factors, it was necessary in this investigation to go back to the depots, to contact contractor technical representatives, and to question them at length to obtain the necessary detailed information. It would be helpful to engineering progress in reliability and maintainability if data collection systems would yield more descriptive material and illustrations or photographs of the failures experienced. It is recognized that this recommendation would be countered by the argument that such a system would be costly. It may well be that it is more economical to continue with the procedure used in this investigation, namely, to use the data bank only to highlight areas needing further investigation and then to proceed with specific detailed questions.

The Failure Mode Effect and Reliability Analysis as carried out in Appendix II showed reasonable correlation with the field reported failures as tabulated in Table XIII as follows:

Main rotor pylon hinged cover - five modes identified by FMERA vs 11 field reported or 45%.

Lower personnel door - 12 modes identified by FMERA vs 18 field reported or 67%.

Main rotor pylon work platform - five modes identified by FMERA vs six modes reported by the field or 83%.

The results indicate that the FMERA is a useful tool in combination with other techniques such as use of reliability/maintainability data bank, but that it is not adequate to be used exclusively.

The trade-off analysis technique is useful in determining the cost effectiveness of reliability and maintainability improvements.

Among other reliability and maintainability techniques not covered by this investigation that are believed to be useful are: prediction and allocation, design reviews, tracking and measuring reliability during development

math modeling, and time line analyses.

Design and Test Criteria: Recommended Revisions

The recommendations for specification changes to improve secondary structures were shown to be valid by the test results. There were deficiencies in design, most of which could be prevented by the specification revisions.

Note, however, that the addition to MIL-T-8679, paragraph 3.1.10.7, includes a test loading table that is not universal. These loads reflect the use cycles peculiar to the H-53 aircraft. Another model helicopter would be subject to different loadings, due to the differences in passenger and crew capacities and the different maintenance requirements.

SD-24H, Volume II, although used in the design of some of the components under study, has been superseded by SD-24J, Volume II. Since comments on SH-24H have been covered under SD-24J, the former has been dropped from further consideration.

SD-24J, Volume II, paragraphs 3.2.4.2.4 and 3.11.7 should be revised as follows:

3.2.4.2.4 DOORS, MOVABLE SECTIONS, AND REMOVABLE SECTIONS. - Doors, movable sections, or removable sections, shall be provided for inspection, lubrication, servicing of engine, transmission, rotor head, and accessories, drainage, removal of corrosion deposits, adjustment, refinishing, and replacement of parts as required. Doors, movable sections, and removable sections shall furnish an adequate view of the parts to be inspected and provide ample access to parts involved to permit disconnection and removal of a part without having to remove other parts or units not affected. Doors, movable sections, and removable sections shall be suitably identified. Doors shall be externally smooth, splashtight, readily opened, securely closed and may employ transparent windows subject to Government approval. Doors shall be designed to prevent damage due to airblast, shall be hinged on forward or upper edges, where practicable, and shall be capable of withstanding all combinations of pressure distribution and accelerations resulting from the specified rotary-wing aircraft design conditions. Load-carrying doors, movable sections, and removable sections shall not be used where removal is necessary for periodic inspection, but otherwise may be used where weight savings result. Threaded-tapered fasteners or other compensating assembly devices shall be used to simplify assembly and reduce maintenance on load-carrying doors, movable sections, or removable sections subject to extreme temperature variations and resultant thermal distortions. Doors, movable sections, and removable sections which must be removed for periodic inspections shall be secured by readily-operated approved flush-type fasteners of corrosion-resistant material. These fasteners shall be common to all doors to the maximum extent practicable, shall be either captive or of identical length, grip, thread and material and shall conform to Spec MIL-F-5591 where applicable. Doors, or movable sections which are required to be held open for a period of time to permit access for maintenance purposes, shall be capable of being secured in both the open and closed positions by self-locking devices. However, where no useful purpose is served,

the use of a device for securing a door or movable section in the open position is not required. If higher performance characteristics than those of Spec MIL-F-5591 are required, quick acting rotary fasteners conforming to Spec MIL-F-22978 shall be used. Contact areas between doors and aircraft structure shall be protected against fretting corrosion by providing suitable insulating materials.

Rationale:

Addition of the words "movable sections," "transmission," and "rotor head" would identify these items as pertinent to helicopters and within the scope of these requirements.

3.11.7 Integral Work~~ing~~ Platforms. - Integral ~~engine~~ work~~ing~~ platforms ~~for engine maintenance which cannot be readily performed from the ground (deck),~~ shall be provided~~,~~ to permit access to and maintenance of engines, transmissions, and rotor heads which cannot be reached readily from other parts of the aircraft, the ground, or the ship's deck.

Rationale:

Addition of transmissions and rotor heads would identify those items as pertinent to helicopters and within the scope of these requirements.

MIL-T-8679, paragraph 3.1.10.7 should be revised to read as follows:

3.1.10.7 Deformation and fatigue of doors, work platforms, movable or removable covering or fairing, cowlings, locks, latches, slides, rollers, and fasteners. - It shall be shown during structural tests that ~~doors, cowlings, movable and removable coverings,~~ these items and items of mechanical equipment, such as landing gears, remain in their intended positions consistent with specified structural design requirements. It shall also be shown that the following fatigue or repeated load tests have been met:

Item	Open/Close Cycles	Repeated Impact Force	Repeated Impact Cycles
Door Entrance			
a) with stairs	1,000	200 lb x(man rating)	20,000
b) without stairs	1,000	Slamming	1,000
Door Inspection			
a) hinged	100	Slamming	1,000
Platforms, Work			
a) operable	1,000	200 lb x(man rating)	20,000
b) fixed	-	Same	20,000
Cowling, Covering, Fairing			

a) removable	1,000	Slamming/Drop - 5
b) hinged	1,000	Full Slam/Drop - 5
c) sliding	1,000	Slamming/Drop - 5

Rationale:

This proposed revision would provide specific requirements for testing secondary structure items under conditions simulating actual service operations and abuse. Although these test conditions may not apply universally because of differing conditions of loadings, passenger and crew capacities, and maintenance requirements applicable to various helicopters, they do form a base.

MIL-S-8698(ASG)(-1), paragraphs 3.1.3.3 and 3.2.2.2 should be revised to read as follows:

3.1.3.3 Doors, cowling, <u>integral work platforms, movable or removable covering or fairing,</u> locks, and fasteners. - Doors, cowling, <u>integral work platforms, movable or removable covering or fairing,</u> locks, and fasteners, including landing gear up and down locks and cowling fasteners, shall not deflect from their intended positions in such manner as to permit unwanted openings, closing, or release of coverings, or unlocking or unfastening of mechanisms at all loads up to ultimate.

3.2.2.2 <u>Design fatigue loading.</u> - The design fatigue loading shall be in accordance with an approved fatigue design loading schedule. The helicopter and its components, except those covered by applicable specifications, shall be designed for a minimum fatigue life of 1,000 hours. <u>Design fatigue loading for doors, boarding steps, integral work platforms, and movable or removable covering or fairing shall include loads and effects of abuse (slamming, jumping, kicking, forcing, etc) imposed by personnel during loading, boarding, inspection, and maintenance of the aircraft.</u>

Rationale for 3.1.3.3 and 3.2.2.2 improvements:

The additions (underlined) above would cover heretofore unspecified secondary structure requirements.

MIL-I-83294(USAF), paragraph 3.4.9.5. Although this paragraph was considered for expansion to include fairings, cowling, and integral work platform requirements for access to transmissions and rotor heads, further study of the entire specification indicated that this would be impractical. This specification is apparently intended for application to fixed-wing aircraft propulsion (engine installation) systems; it does not include even the basic requirements for helicopter type propulsion transmission systems (gearboxes, shafting, rotor heads, etc.).

AFSC DH 2-1, DN 2A1, paragraph 4 should be revised as follows:

4. DYNAMIC LOADS

Dynamic loads are time-dependent forces. The application of these forces to the flexible airframe structure usually results in magnification of displacements or stresses in the airframe over that which would have occurred if they were applied statically. Dynamic load effects have been found to be important during the following conditions:

a. Taxiing, takeoff and landing

b. Flight through gusts

c. Gunfiring

d. Rocket accelerated takeoff

e. Abrupt aircraft maneuvers

f. Static engine run-up during maintenance and takeoff

g. Store ejection

h. Operation of doors, work platforms, movable or removable covering or fairing, cowlings, etc. during loading, boarding, inspection, and maintenance operations; as applicable.

Design the airframe structure so that the magnitudes and distributions of loads include the dynamic response of the structure resulting from the transient or sudden application of loads. Conform to the specific design and load test requirements stated in MIL-A-8860 through MIL-A-8871. (See 1)

Rationale:

Addition of subparagraph "h" would direct attention to a heretofore neglected area. Addition of "(See 1)" as last sentence would direct user to separate helicopter requirements.

AFSC DH 2-1, DN 3A3, paragraph 3 should be revised as follows:

3. FAIRING

Fairing is covering which may or may not be an integral part of the aircraft structure and whose primary purpose is to increase the aerodynamic efficiency of the aircraft. Protect loose edges on fairing by adequate rubbing strips. Construct fairings in the same manner as covering or cowling. If seldom removed, fairings may be screwed or bolted to the adjacent covering or structure. When fairing is secured with threaded fasteners having the same diameter, use bolts or screws of equal length to simplify maintenance. If frequently removed, provide suitable cowling fasteners. If removal for inspection of disassembly is not necessary,

fairing may be fastened by riveting or welding. <u>If fairing is of the sliding type, it should meet the deformation and fatigue requirements of MIL-T-8679, paragraph 3.1.10.7.</u>

Rationale:

Addition of the last sentence would direct attention to a heretofore unrecognized area.

AFSC DH 2-1, DN 3A3, new paragraph 9 should be added as follows:

9. <u>INTEGRAL WORK PLATFORM (HELICOPTER)</u>

<u>Integral work platforms shall be provided to permit access to and maintenance of engines, transmissions, and rotor heads which cannot be reached readily from other parts of the aircraft, the ground, or ship's deck.</u>

Rationale:

Addition of the above would cover a heretofore unspecified requirement.

AR-56, paragraph 3.1.9.1 should be revised to read as follows:

3.1.9.1 <u>Design Fatigue Loading.</u> - The design fatigue loading shall be in accordance with an approved fatigue design loading schedule based on realistic mission profiles or in accordance with the profile(s) of Table I. These profiles shall be combined with a rational distribution of significant parameters which affect fatigue life, including cg, altitude, gross weight, load factor/bank angle, yaw angle, sinking speed, roll angle, pitch angle, takeoff-landing speeds, soil conditions, rotor speeds, rotor-hub moments, control loads, torque variations, vibratory loadings, quasi-static loads, landing gear extension-retraction loads and all others pertinent to describing the fatigue loading spectra that the vehicle will be subjected to. Safe life analyses and tests shall be employed to substantiate the helicopter and all its components for a fatigue life specified in 3.1.9.2. <u>Design fatigue loading for doors, boarding steps, integral work platforms, and movable or removable covering or fairing shall include loads and effects of abuse (slamming, jumping, kicking, forcing, etc.) imposed by personnel during loading, boarding, inspection, and maintenance of the aircraft.</u>

Rationale:

The addition (underlined) above would cover heretofore unspecified secondary structure requirements.

SS 9583. This Sikorsky specification, previously approved by the Government, is being revised to require materials with improved interlaminar shear strength not currently provided in military specification materials. We will request that the military specification also be revised in the near future. This should reduce delamination problems to a minimum. However, in order to eliminate delamination and cracking of fiber glass, good judgment

must be exercised in designs which are subject to penetration (dropped tools, hard heels, etc.) and localized pounding (due to vibration or repeated loads resulting from normal looseness of latches and other fasteners) to determine if fiber glass is suitable for the application, if metal reinforcement is required, or if metal should be used instead. Fiber glass might delaminate or crack under such conditions; metal would probably yield instead. (We recommend that the Army include such design information in AMCP 706-202 when issued.)

TEST PROGRAM

Background

Tests of the original and redesigned structures were conducted to:

(a) duplicate the in-service inadequacies of the original designs and

(b) demonstrate improvement of the redesigned structure.

The problem areas of the three selected components were reviewed to determine the service conditions (loading spectra, vibration and aerodynamic environment, and abuse) which most likely contributed to the in-service failure modes of each part. Laboratory tests were designed to integrate these conditions into a combined test spectrum designed to duplicate these in-service modes. Scheduled usage and estimates of nonscheduled usage and abuse were employed to combine the individual service conditions into composite test programs that permit interaction of these conditions in proportion to field exposure. No attempt was made to quantify the reliability of present or redesigned parts with these spectra. However, based on the results of the limited testing conducted, an estimate of the improvement in reliability which will result from the redesigns has been made in the preliminary trade-off study.

Criteria

In order to develop a realistic test of the secondary structures, several assumptions were made concerning aircraft use (Table VIII).

Using these criteria, a test schedule was set up, cycling the structures through manual operation, abuse, flight loads, and normal use as well as environmental testing (Table IX).

Loads for these phases were established as described in the following paragraphs.

TABLE VIII. TEST CRITERIA ASSUMPTIONS

General

(1) 1½ hours per flight (scheduled)

(2) average loading (3/4 x 40 = 30 passengers estimated using personnel door)

(3) preflight inspection (scheduled - 1 man)

(4) postflight inspection (scheduled - 1 man)

(5) 1 maintenance per flight (2 men estimated)

Specific

Main Rotor Pylon Hinged Cover Assembly - 3 open/close cycles per flight hour

Lower Personnel Door - 3 entries/exits per 1½ hour scheduled flight

Work Platform - 2-man rating

Environmental

Humid, salty air environment simulated by application of 5% salt-water solution as suggested in MIL-STD-810.

Test Load Schedules

Set up to produce interaction between test phases

TABLE IX. TEST LOAD SCHEDULES - 24-HOUR BLOCKS

		Test Total
Main Rotor Pylon Hinged Cover Assembly		(25 Blocks)
(1)	60 open/close cycles, 20 of which are abusive	1500 cyc
(2)	20 hours vibratory load	500 hr
(3)	Functional check-out	
Lower Personnel Door		(32 Blocks)
(1)	Functional check-out	
(2)	Salt water spray (apply periodically)	
(3)	32 open/close cycles	1024
(4)	720 stair tread impacts	23000
(5)	160 stair riser impacts	5100
(6)	32 support cable impacts	1024
(7)	16 hours vibratory load	500 hr
Work Platform		(25 Blocks)
(1)	Functional check-out	
(2)	40 open/close cycles, 10 of which are abusive	1000
(3)	160 roller cycles	4000
(4)	20 vibratory load	500 hr
(5)	Salt water spray (apply periodically)	

Flight Load Study

To set up the vibratory test load parameters, an aircraft was instrumented to provide actual flight data. Strain gages were fixed to the flight secondary structures at critical points (near latches, hinges, and stress concentration points - if any); see Figures 1, 2, 3 and 4. The results, Table X, indicate that flight stresses are low. This data was gathered during the Sikorsky Aircraft RH-53D flight test program in February, 1973.

Vibratory Loading

The vibratory loads were induced on the test articles in an attempt to simulate flight vibrations. This was found to be difficult. It was not possible to match the low levels of inflight stress (Table X) recorded at the data points (strain gages) without creating extremely high localized loadings at the load input points. It was noted that the deflection of the test article was not directly proportional to the indicated stress at the test data points.

To get more than a localized load, it was necessary to operate at or near a resonance frequency of the system. This method produced measurable deflections and loads in the test items. However, they were still not comparable to flight test data in distribution of magnitude.

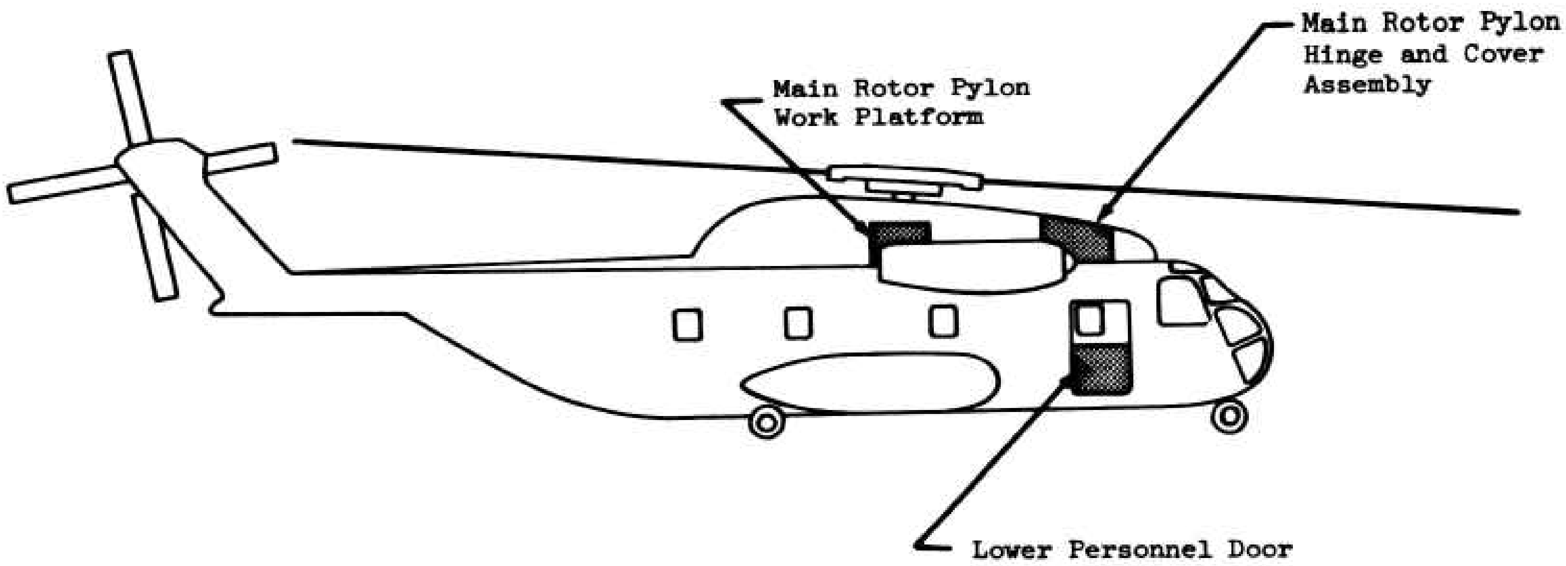


Figure 1. S-65 Secondary Structure Test Articles.

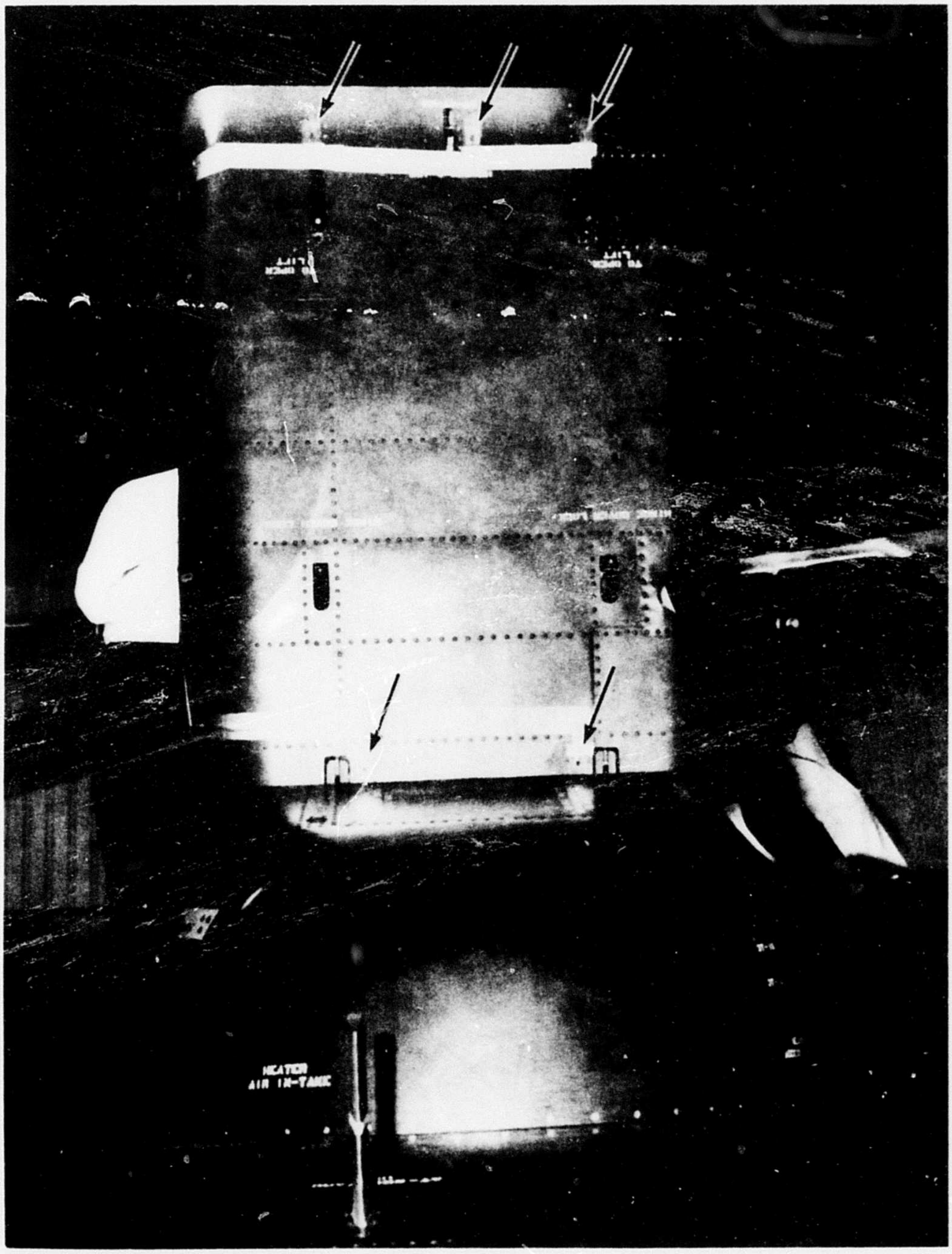


Figure 2. Flight Vibratory Load Survey, Strain Gage Locations, Hinged Cover.

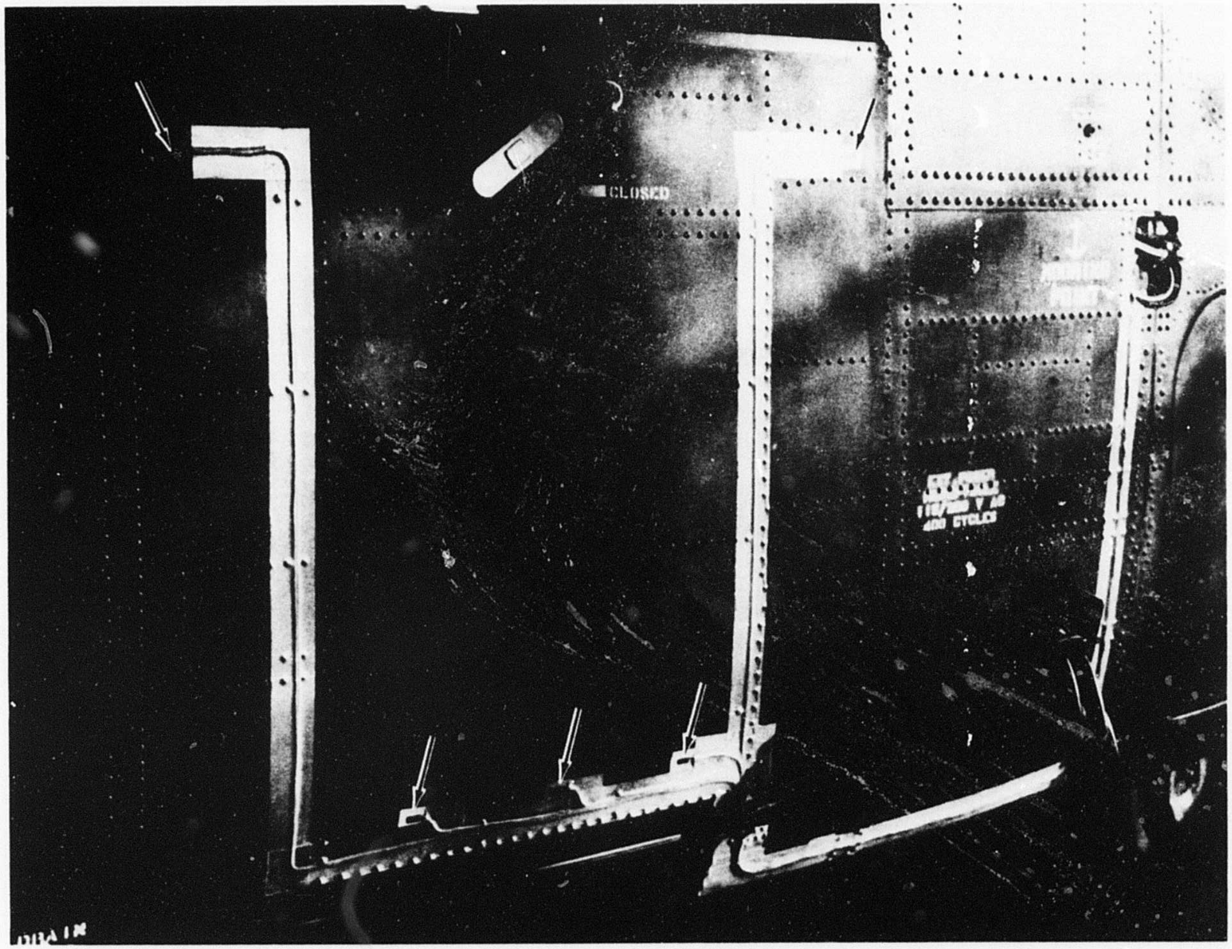


Figure 3. Flight Vibratory Load Survey, Strain Gage Locations, Personnel Door.

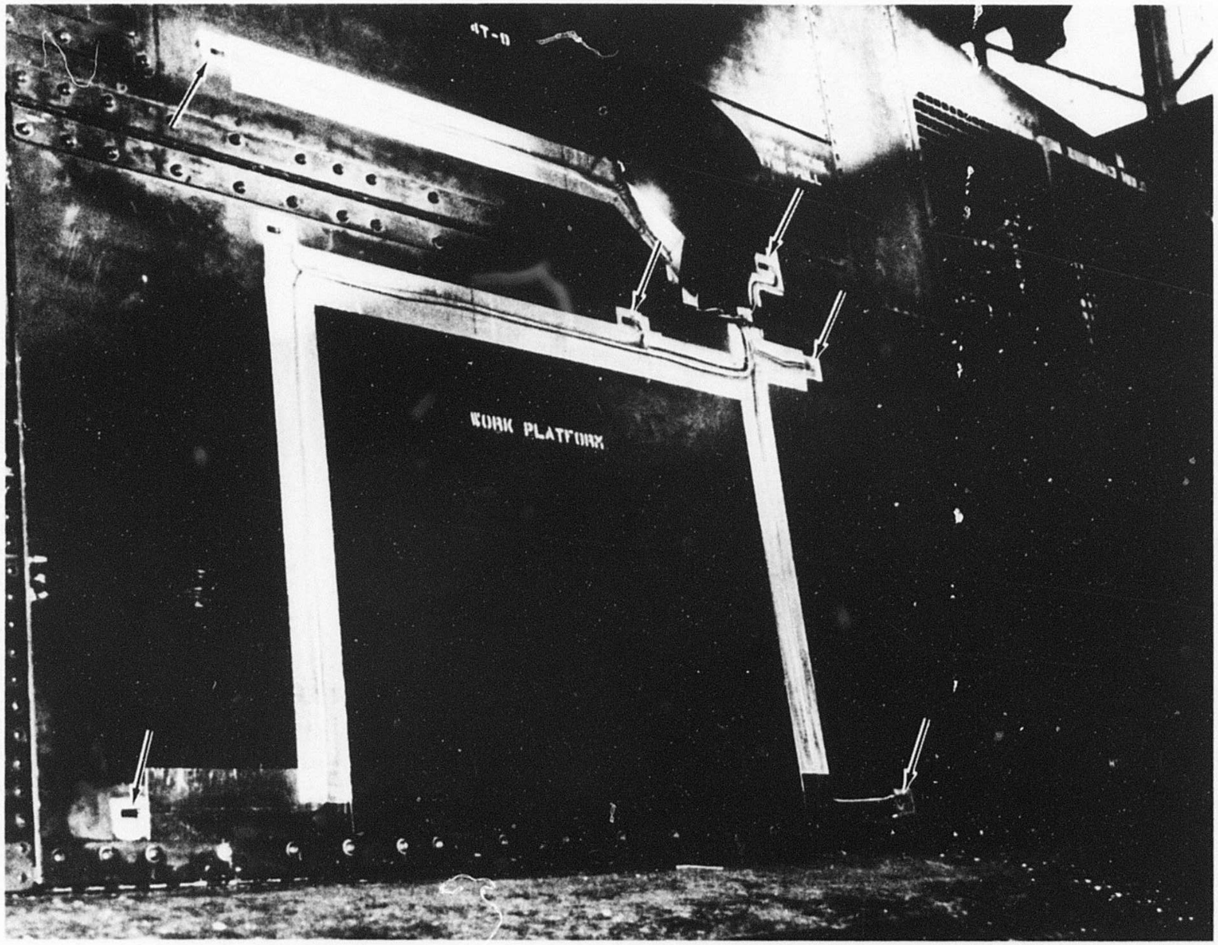


Figure 4. Flight Vibratory Load Survey, Strain Gage Locations, Work Platform.

TABLE X. FLIGHT STRESS AND VIBRATION STRAIN GAGE DATA (MAXIMUMS)

Test Article	Stresses (psi)	Frequency
Hinged Pylon Cover-Gages		
PC-1	250	All Approximately 6/Main Rotor (185 Rotor RPM) (6 Blades/Rotor)/(60 $\frac{sec}{min}$) = 18.5 cps
PC-2	400	
PC-3	200	@ 18.5 cps
PC-4	150	
PC-5	200	
Personnel Door-Gages		
PD-1	200	Same
PD-2	150	
PD-3	100	@ 18.5 cps
PD-4	450	
PD-5	Out	
Work Platform-Gages		
WP-1	600	Same
WP-2	500	@ 18.5 cps
WP-3	200	
WP-4	350	
WP-5	250	
WP-6	200	
WP-7	200	

Because of the problems involved, each original design test article was run at a reasonable load and frequency level, which was duplicated on the redesigned item to provide the design comparison (Table XI).

The devices used to set up the vibratory load test parameters were:

(1) Counter (Figure 5, top) - to indicate cycles per second.

(2) Load Monitor (Figure 5, middle) - to monitor a master strain gage on the test article and shut down the equipment if the load increased or decreased by more than a preset percentage.

(3) Power Panel (Figure 5, bottom) - to indicate the total running time of the vibratory load.

(4) Oscillograph - to read out the strain on the test articles.

The vibratory load was induced by a variable-speed motor driving a wheel with an eccentric weight (Figure 6). The wheel and carriage assembly actuated a pushrod attached to a frame mounted on rubber pads bonded to the test item (Figure 7).

TABLE XI. VIBRATORY TEST RESULTS

Structure	CPS	Input Force (lb)	Comments
Pylon Cover - Original	15	±29	Vibratory load not run on redesigned structure due to structural damage that occurred on original - Not Field Mode of Damage.
Pylon Cover - Redesign	Not Run		
Personnel Door - Original Personnel Door - Redesign	22	±41	Field Damage Modes Developed.
Work Platform - Original Work Platform - Redesign	13	±25	No Damage Modes Developed.

Environmental Test

To simulate a high-humidity, salty, ocean-air environment, a salt water solution (5% as suggested in MIL-STD-810) was applied to appropriate points (hinges and latches) of the test articles. However, the test was too short (one month, to simulate 500 hours of flight time) to realistically

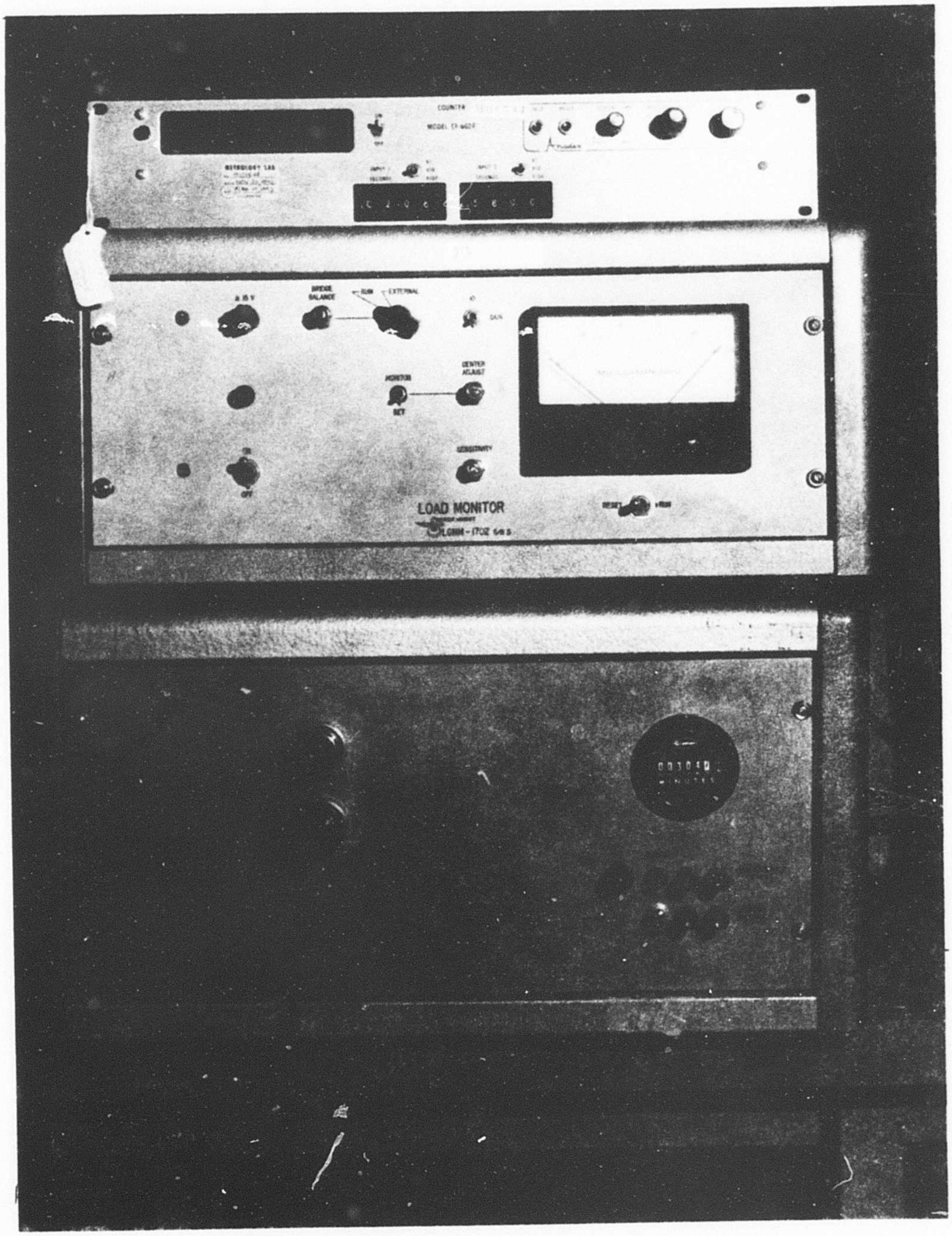


Figure 5. Vibratory Load, Instrumentation.

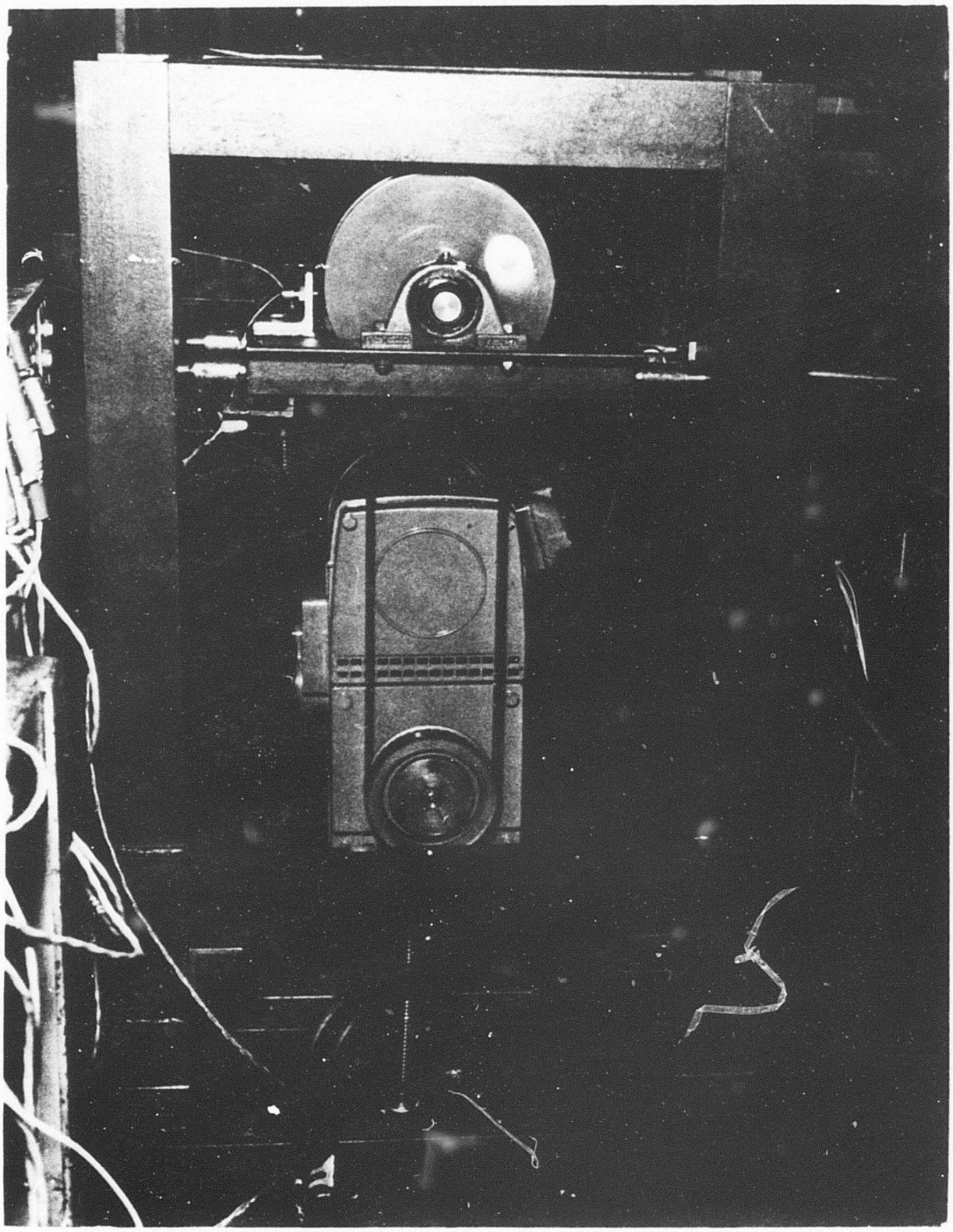


Figure 6. Vibratory Load, Varidrive Motor and Wheel and Carriage Assembly.

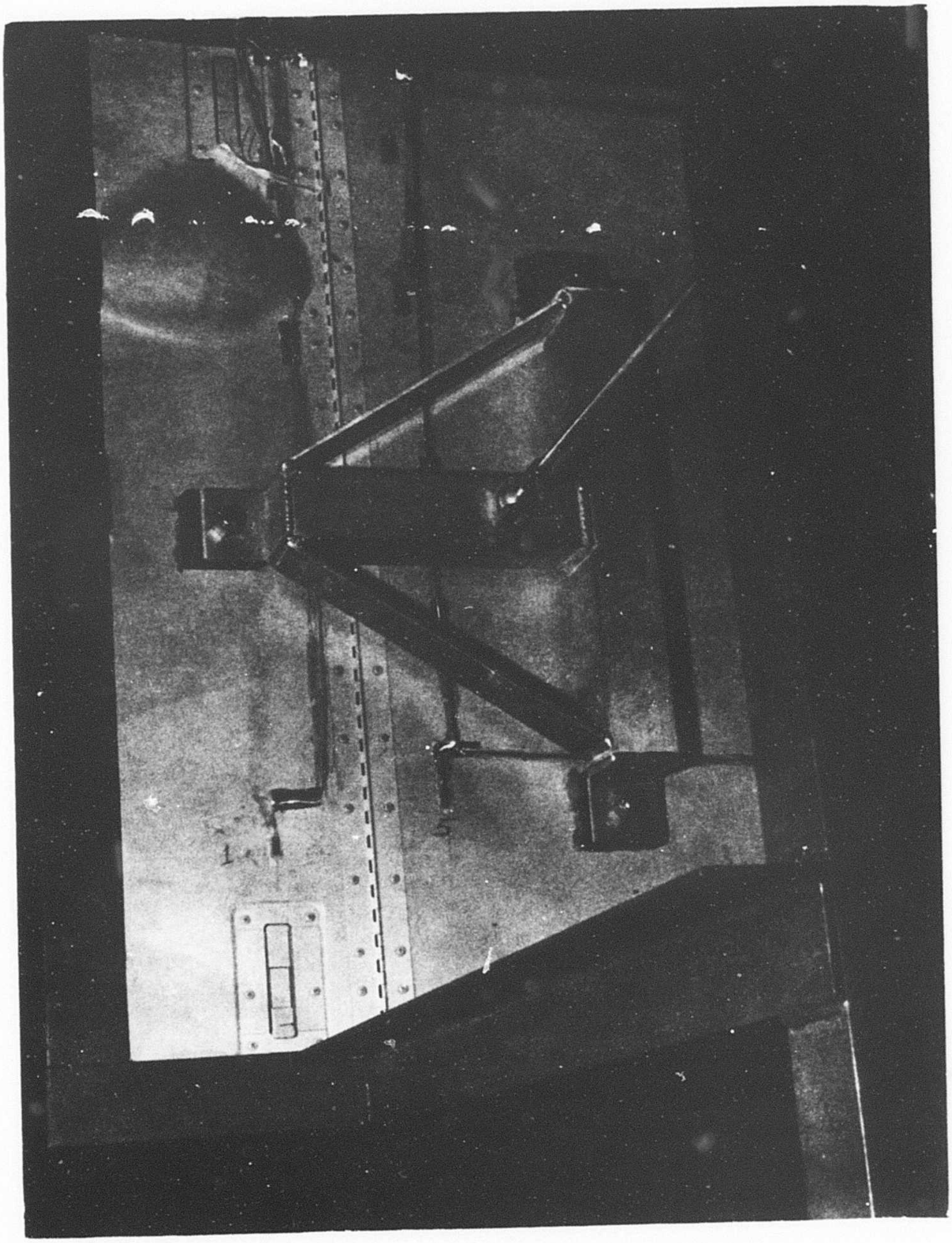


Figure 7. Vibratory Load, Load Frame.

evaluate corrosion problems, and none were observed.

Field Abuse

Field abuse of secondary structures involves the human engineering aspect of design and the mental attitude of field personnel. Abuses observed in the field such as kicking, jumping on, and slamming of secondary structures can cause loadings (and damage) for which no design provision was made.

Abusive damage can be invited by (1) difficulty in the normal operation of the secondary structure such that it must be forced to operate, (2) design of the secondary structure lending itself to a function never intended by design, or (3) a secondary structure so fragile that even normal use can cause damage. See Table XII for apparent and observed abuses.

The amount of abuse scheduled in the test block diagrams was arrived at by estimates of field usage.

Field Damage

Field damage reports are difficult to assess. They are often composed from incomplete maintenance and work data that do not specify the actual problem areas. The common terms used, such as "cracking, bending, delaminating," give only a vague indication of what happened.

Superior information has been obtained from Sikorsky field representatives at CH-53 bases. On request, they have provided specific information on damage - part number, type of damage, and probable cause. This is the type of data needed to redesign and retest effectively.

The field modes of damage and those duplicated in the course of testing are given in Table XIII.

TABLE XII. FIELD ABUSE - ORIGINAL DESIGNS

Structure	Abuse	Reason	Attributed Damage
Pylon Cover	Dropping Open	Carelessness, Wind Gust Blew Open	Bending, & Buckling of Structure - Misalignment
	Dropping Closed	Same	Same
	Slamming Shut	Cover Will Not Close due to Misalignments or Interference	Same
	Forcing Latches	Latches Will Not Close due to Misalignment, Poor Linkage Design	Latch Mechanism Breaking
	Kicking Sides Of Cover	To Get Cover to Align & Close	Breaking, Bending, & Cracking
L. Personnel Door	Slamming Shut	Hard to Close due to Edge Seal Binding	Door Edge Cracks
	Forcing Latch Assembly	Same	Broken Latch Mechanism
	Dropping Open	Easiest Way to Lower Door	Elongation of Holes, Cracks of Support Installation
	Stepping On Cable	Quickest Path off Sponson Work Platform	Wearing, Cracking, Breaking of Support Assembly Installation
	Kicking Stair Riser	Riser Exposed to Boot Contact	Cracking of Riser

TABLE XII. - Concluded

Structure	Abuse	Reason	Attributed Damage
Work Platform	Slamming Shut	Binding of Hinges, Interference of Edge Seals on Aircraft Structure	General Chipping, Cracking, & Fraying of Edges and Rubber Seals
	Dropping Open	Carelessness	

TABLE XIII. DAMAGE MODES OBSERVED IN THE FIELD AND DUPLICATED IN TEST

Main Rotor Pylon Hinged Cover Assembly Damage	Field	Test Original	Redesign
(1) Structural Damage:			
a) Frames Bent and Cracked	X	X	
b) Skin Bent and Cracked	X	X	
c) Fwd Lower Corners - Cracking, Fraying	X	X	
d) Buckling of Frames - Pulled Rivets	X	X	
e) Lower Aft Corners - Chafing, Bending	X	X	X
(2) APP Exhaust Burns	X		
(3) Latch Malfunctions:			
a) Wear and Tear	X	X	
b) Bending, Breaking of Mechanism and Parts	X		
c) Shearing Rivets	X	X	
d) Opened - Fwd Locking Pins - Misaligned, Bending, Breaking	X	X	
e) General Misalignment - Difficulty in Operation	X	X	*

Lower Personnel Door Damage	Field	Test Original	Redesign
(1) Door:			
a) Steps Cracking	X		
b) Exterior Skin Cracks	X	X	*
c) Distortion	X	X	
d) Dents	X		
e) Bending	X	X	
f) Misalignment	X	X	X
(2) Support Assembly:			
a) Wear at Attachment Points	X	X	
b) Cracking	X		
(3) Latch:			
a) Distortion	X	X	
b) Wear	X	X	
c) Breaking	X		
(4) Hinge: Cracking	X		

TABLE XIII - Concluded

Lower Personnel Door	Test		
Damage	Field	Original	Redesign
(5) Cables: Breaking, Fraying	X		
(6) Structural Damage:			
a) Center Step Riser	X	X	*
b) Latch Assembly Support Channels	X	X	*
c) Side Beam Step Support Cracks	X	X	X
(7) Door Seal:			
a) Binding	X	X	
b) Damaged - Needing Replacement	X		
Main Rotor Pylon Work Platform Assembly	**Test**		
Damage	Field	Original	Redesign
(1) Delamination - Fiber Glass	X		
(2) Distortion	X		
(3) Latch Assembly - Breaking	X		
(4) Hinges:			
a) Halves Breaking	X		
b) Pins Breaking and Working Loose	X		
(5) Cracking	X	X	

*Although damage occurred, the redesigned exhibits improved performance.

REDESIGN

Field maintenance data and test results were used to develop redesigns of the three secondary structures.

The rework of the pylon cover was based on the results generated in the secondary structures test. This was possible due to the shorter lead time to rework an existing cover compared with redesigning whole components, as was done for the work platform and the lower personnel door (Table XIV).

The redesign of the lower personnel door was initiated on the basis of field maintenance information from Sikorsky field representatives at the CH-53 base on North Island, San Diego, California (Table XV).

TABLE XIV. REDESIGN OF HINGED COVER ASSEMBLY (P/N 65205-09010-011)

Damage	Probable Cause	Corrective Action
(1) Frames & Skin Bent & Cracked - General Damage	Flexibility Allows Misalignment	a) Two Rows of Intercostals Added to Side Panels to Increase Panel Stiffness
(2) Interference of Forward Lower Corner With Alignment Pin Striker Assembly	Flexibility of Cover	a) Same as Item 1 b) Cut Interfering Flange 90° to Outer Contour (Currently @ 45°)
(3) Forward Open Alignment Pin Forced Out of Housing	Shock Load During Free-Fall Drop Test	Added Retaining Channel to Contain Pin Within Housing
(4) Middle Latch Pin Bent	Sticks Out During Free-Fall	Added a Returning Spring to Fully Retract Pin
(5) Main Latches a) Hard to Operate b) Rivets on Operating Lever - Shearing	 a) Linkage Overcenters b) Not Enough Rivets	 a) Reduced Length of Slot b) Increased Length of P/N 65209-09025-103 Arm to Hold Additional Rivets
(6) Buckling of Lateral Stringer Between Hinges	Free-Fall Load to Open Position	Reinforced Frame
(7) Buckling - Tearing of Stringers Around Forward Open Positioning Pins	Free-Fall Load to Open Position	Reinforced Local Area

TABLE XV. REDESIGN OF LOWER PERSONNEL DOOR (P/N 65207-03018-041)

	Damage	Probable Cause	Corrective Action
1)	External Skin Cracks, Distortions	Repetitive Door Closings With Seal Binding on Door Jam, Building up Pressure Between Edge of Door and Jam	a) Redesigned Seal b) Reinforced Door Edge
2)	Cracks in Lower Part of -101 Support	Reduced Cross Section Caused by Cutout for Latching Mechanism	Added Stiffening Angle Tied into Vertical Members at Each End
3)	Cracks in -124 Channel (Middle Step Riser)	Scuffing - Kicking by Personnel	a) Increased Channel Gage b) Reinforced the Attachment at Each End
4)	Cracks in the -102 & -103 Beams	Bend Reliefs in Stiffening Flange and -124 Channel Failure	a) Strapped to Bridge Bend Relief b) Reinforced Channel
5)	Cables Chafing	Cables Rubbing During Open-Close Cycle, and Personnel Stepping on Them	Protective Sleeves Added
6)	Elongated Holes in the -135, -136, -161, & -162 Support Angles	Current Design Allows Bending Moment	a) Redesigned Supports b) Increased Width of the Bushings
7)	Cracks in the -125 Pan	Deflections From Actuation of the Door Handle	Increase Corner Radii to 0.130"
8)	Bend Rods in Latch Assembly	Slam - Lock Feature (of which the rod is part) Does not Function Properly due to Resistance of Door Seal	Same as Item 1.

In the case of the work platform, a design change had already been initiated and introduced into production due to previously noted field failures (Table XVI).

The analysis of the redesigns to prevent the type of failures experienced in the field were qualitative and are included in the description of the changes as follows:

Main Rotor Pylon Hinged Cover

These modifications are shown in detail in Figures 8 and 9.

In zone 17G (Figure 8) is shown the redesigned latching arm with an improved assembly attachment to provide adequate torsional strength.

The balance of the drawing depicts the structural reinforcements shown in Figure 9 for increasing the torsional stiffness of this "U" shaped shell assembly to provide easier handling and improved indexing with mating structure.

Lower Personnel Door

These modifications are shown in detail in Figures 10 and 11.

The new outer skin (-110 in Figure 10) is strengthened by eliminating chem-milling outboard of the stair beams (forward of station 191 and aft of station 213) and maintaining .040 inch thickness to prevent cracking in these areas (see zone 5D and E, Figure 10). The door edges are further strengthened by redesigning the edge members for an improved seal installation (-111 through -116). In the current production configuration, the bulb seal tends to stretch and pinch and strain the adjacent structure, whereas in the redesigned installation, as shown in zone 3A, the seal rolls and deforms. This produces a good seal with a minimum of resistance to contribute to door distortion.

The latch area, shown in zones 5E and 6E in Figure 10, is strengthened by the installation of heavier brackets and clips (-117 through -112).

The redesign to improve strength for resistance to kicking abuse is shown in -101 through -109 (views B-B and C-C), Figure 11.

The redesigned cable support, -043, is shown in detail in Figure 11. The bottom attachment is changed to accept one long through bolt (Figure 30) for a more solid support. The previous installation (Figure 31) had two short bolts which "wracked" from eccentric lug loading. The installation attachment is strengthened (see zone 2B, section D-D, Figure 10) by replacing end rivets with a bolt and radius block.

TABLE XVI. REDESIGN OF WORK PLATFORM ASSEMBLY (P/N 65207-09004-041)

Damage	Probable Cause	Corrective Action
(1) Cracking & Delamination of Fiber Glass	Unknown	Revised Work Platform from Aluminum, Balsa Core, Fiber Glass Laminate to Aluminum Honeycomb Structure
(2) Cracking Around Hinges and Latches	Unknown	

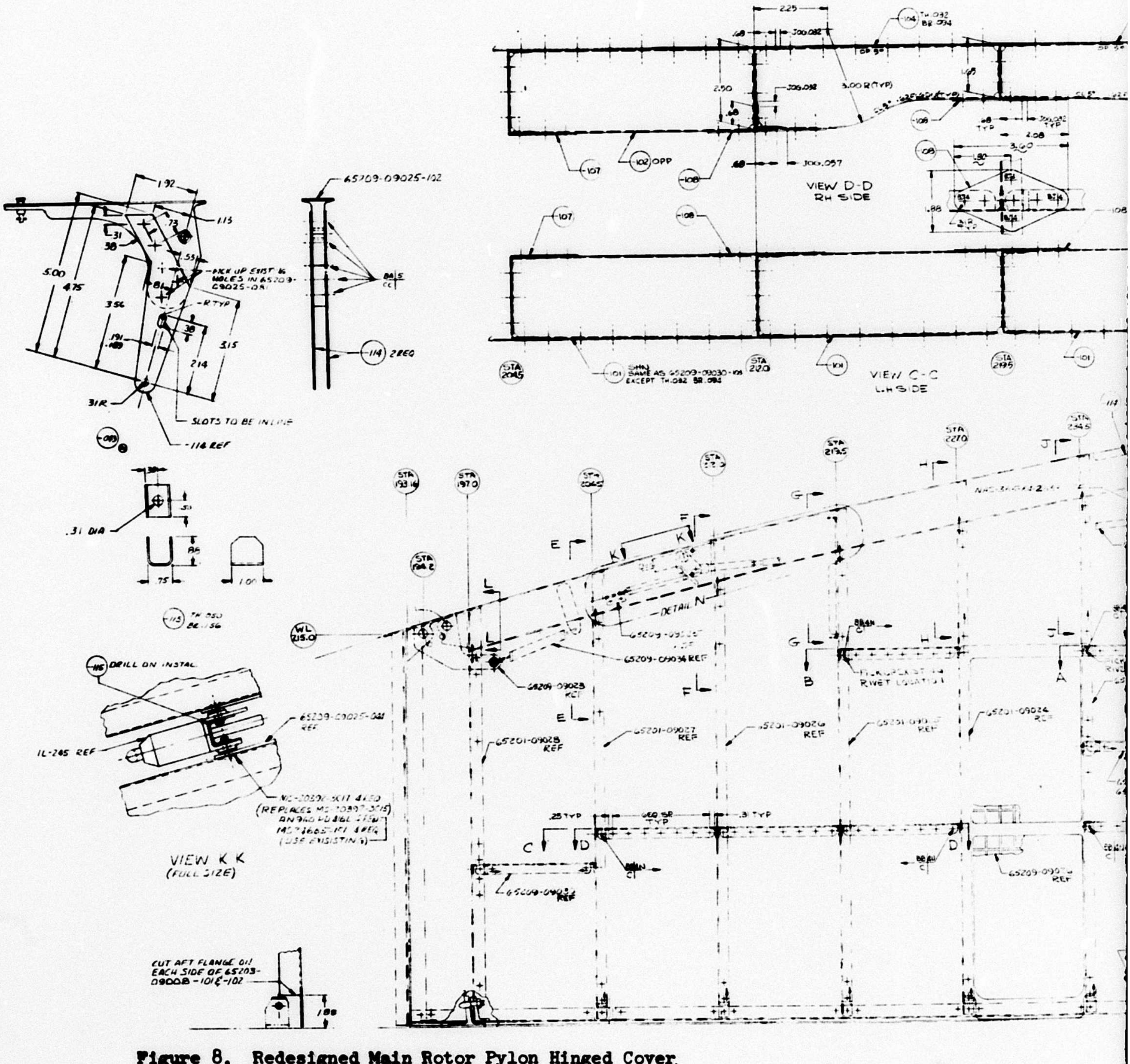
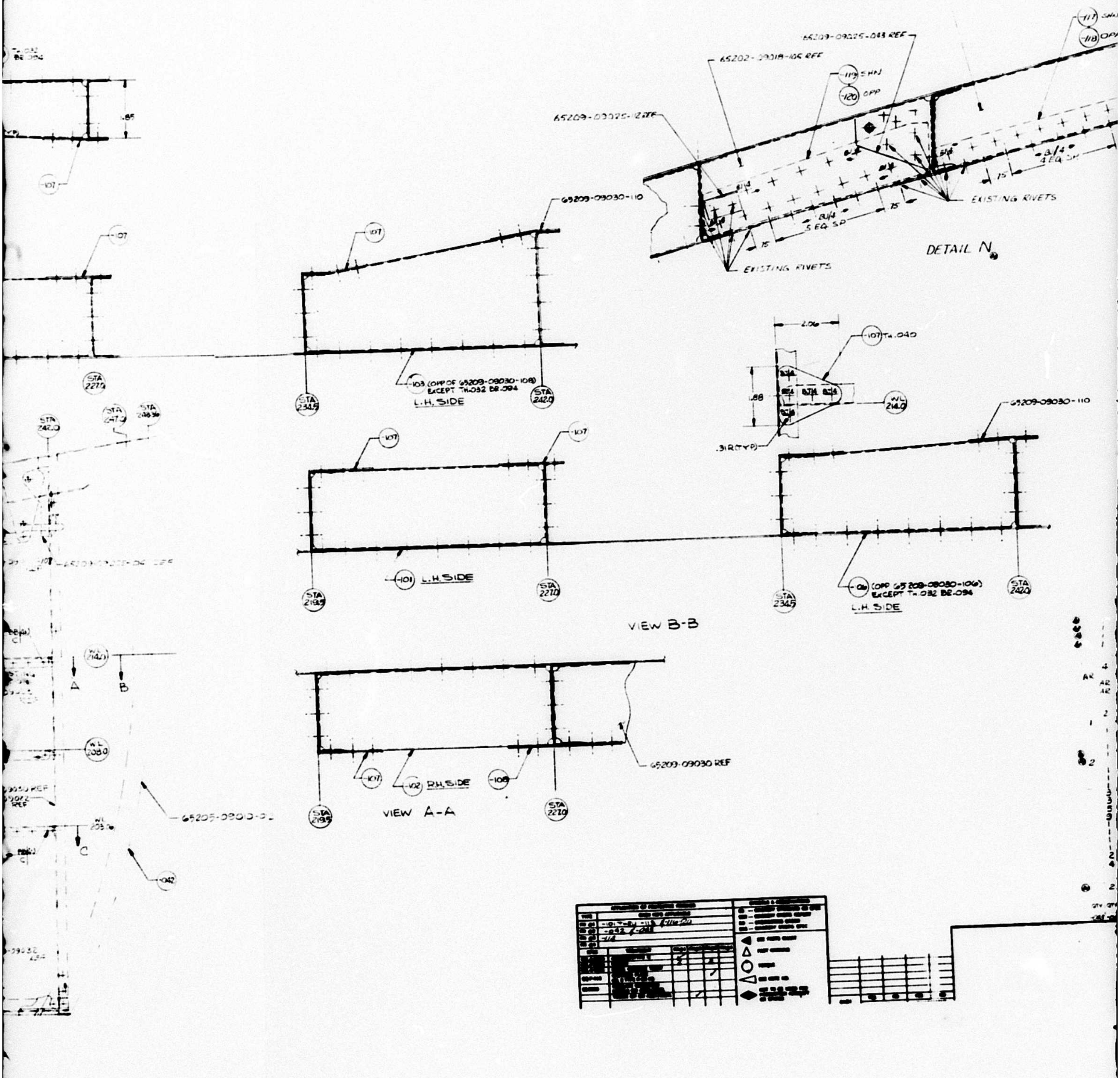
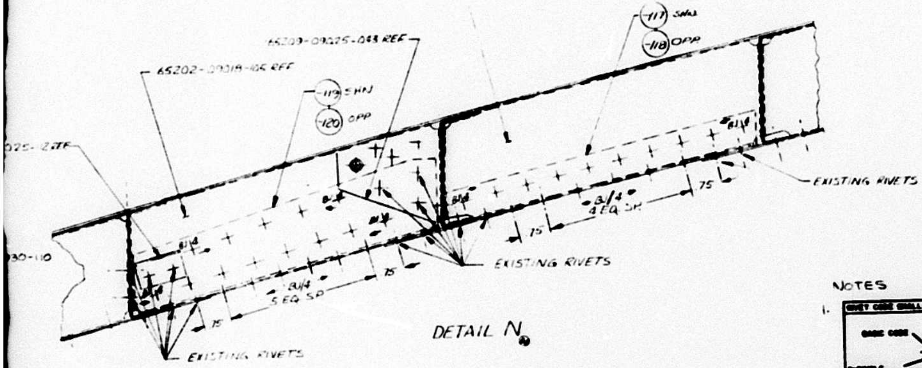
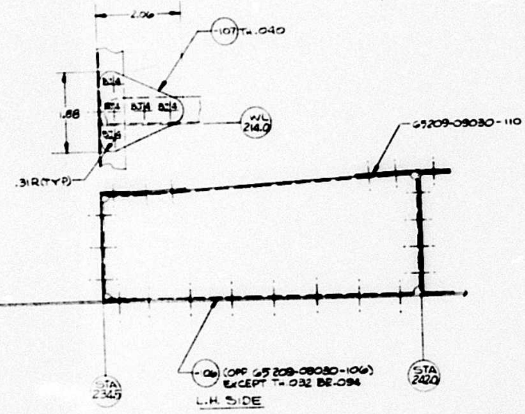


Figure 8. Redesigned Main Rotor Pylon Hinged Cover. (Drawing 38008-20001, Sheet 1).


65209-09025-048 REF
65202-09018-105 REF
-119 SHN
-120 OPP
-117 SHN
-118 OPP
65209-09025-12 REF
8/14
5 EQ SP
4 EQ SP
15
EXISTING RIVETS
EXISTING RIVETS
DETAIL N
1.85
-107
65209-09030-110
-103 (OPP OF 65209-09030-108) EXCEPT TH.032 BE.094
L.H. SIDE
STA 227.0
STA 234.5
STA 242.0
2.06
-107 TH.040
1.88
W.L. 214.0
.31R (TYP)
-101 L.H. SIDE
STA 219.5
-106 (OPP 65209-08080-106) EXCEPT TH.032 BE.094
L.H. SIDE
VIEW B-B
W.L. 214.0
W.L. 208.0
A
B
C
-102 R.H. SIDE
-108
65209-09030 REF
VIEW A-A
-042

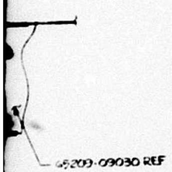


DETAIL N



L.H. SIDE

VIEW B-B



NOTES

1. RIVET CODE

BASIC CODES

2. BREAK ALL SHARP EDGES .005 TO .015R UNLESS OTHERWISE NOTED
3. EXTERNAL CORNER RADII .28 MIN. OR SQUARE OPTIONAL.
4. INTERNAL CORNER RADII .16 MINIMUM.
5. JOGGLES PER SS-1026 AND SS-1027
6. NON-STANDARD ABBREVIATIONS:
   - BR = BEND RADIUS
   - OP = OPEN
   - TR = TRACE
   - TH = TOOLING HOLE
   - = EDGE OF PART
   - CEL = CHEM ETCH LINE
7. AFTER FORMING HEAT TREAT PER MIL-H-6088 TO T42

B. USE IN PLACE OF 65209-09025-081

QTY	QTY	CODE	PART NUMBER	NOMENCLATURE	SPECIFICATION	MATERIAL
1	7		38008-20001-120	ANGLE	QQ-A-250/5	.040 2024-0
1	7		-119	ANGLE		
1	7		-118	ANGLE		
1	7		38008-20001-117	ANGLE	QQ-A-250/5	.040 2024-0
4			MS20392-3C17	PIN		
AR			MS20426-A5	RIVET		
AR			MS20426AD6	RIVET		
AR			MS20470AD4	RIVET		
2			65209-09030-110	GUSSET		
1			65209-09025-102	ARM (REUSE FROM 65209-09025-081 ASSY)		
1			65209-09010-012	COVER & SLIDE ASSY		
	7		38008-20001-116	ANGLE	QQ-A-250/5	.040 2024-0
1	7		-115	CHANNEL	QQ-A-250/5	.040 2024-0
2			-114	TANG	MIL-S-18729	.186A (4130 Cond N
			-113	GUSSET	QQ-A-250/5	.040 2024-0
1	7		-112	OPP		.040
1	7		-111	SHN		.040
1	7		-110	OPP		.040
3	7		-109	SHN		.040 2024-0
3	7		-108			.040 2024-T3
5			-107	GUSSET		.040 2024-T3
9			-106	INTERCOSTAL		.032 2024-0
1	7		-105			.032
1	7		-104			.032
1	7		-103			.032
1	7		-102			.032
2	7		-101	INTERCOSTAL	QQ-A-250/5	.032 2024-0
4	7					
2	8		38008-20001-001	ARM ASSY / COVER & SLIDE ASSY		

QTY QTY SYMBOLS CODE IDENT NO. PART NUMBER

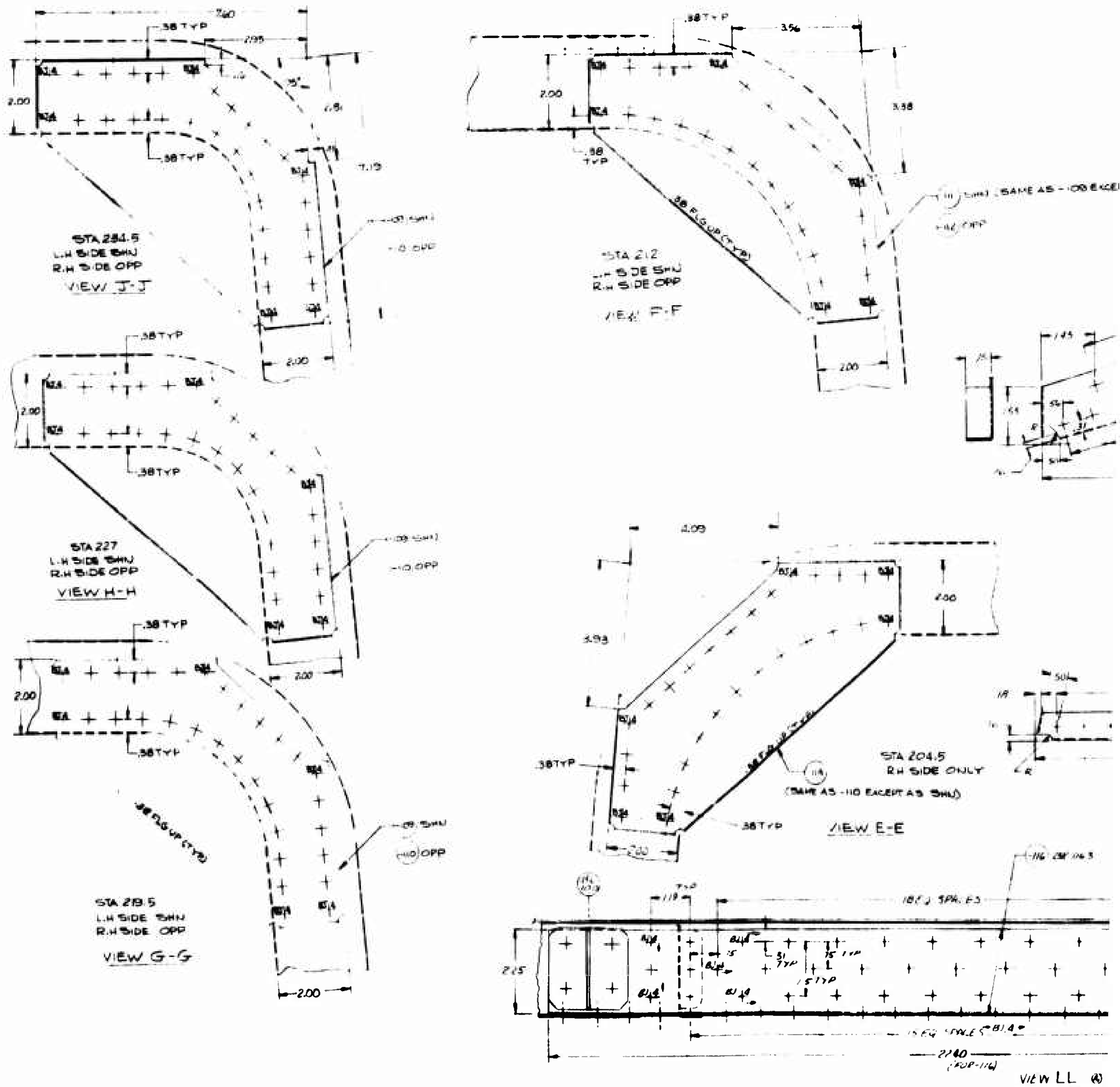


Figure 9. Redesigned Main Rotor Pylon Hinged Cover.
(Drawing 38008-20001, Sheet 2).

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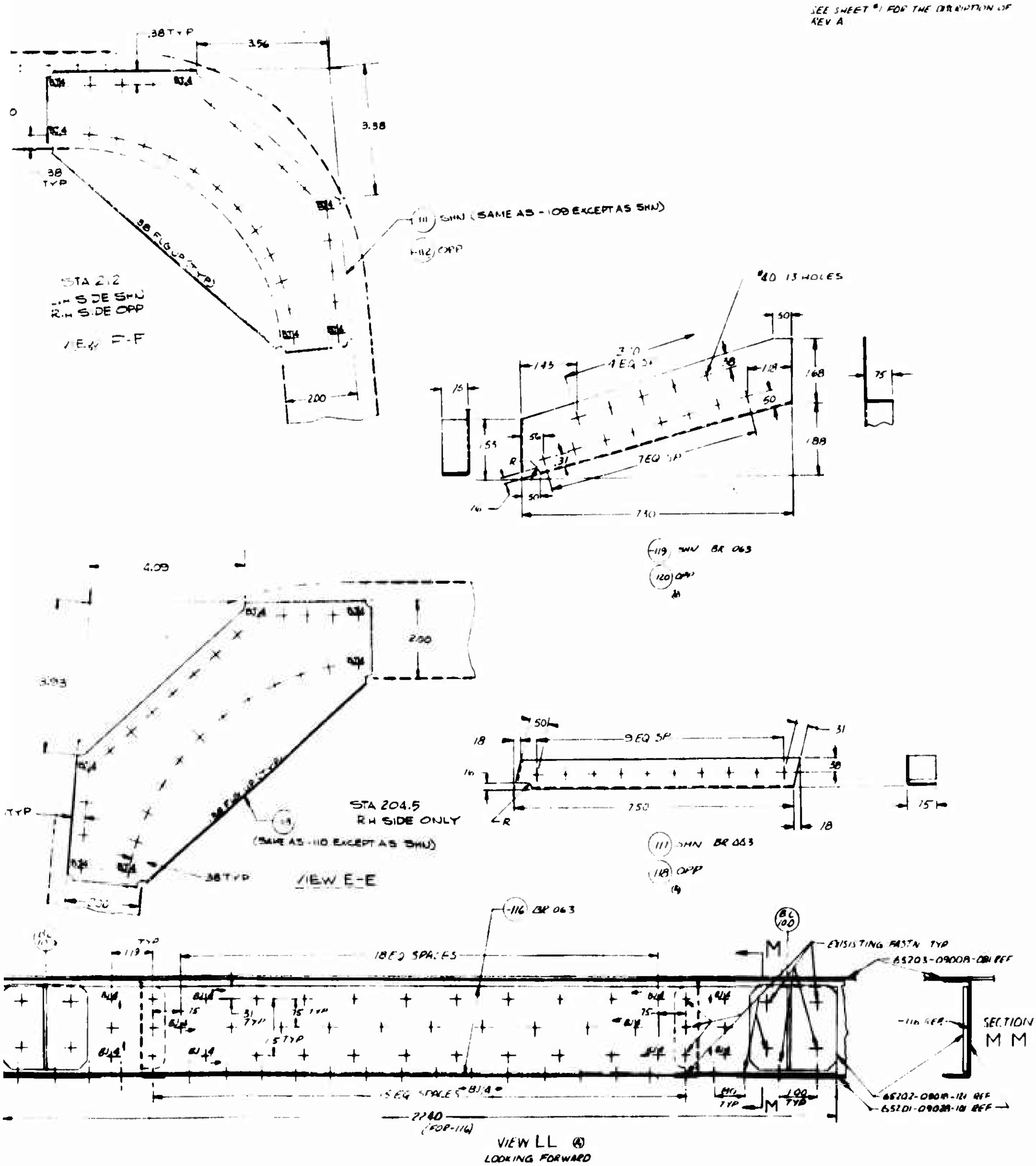

lon Hinged Cover.
eet 2).

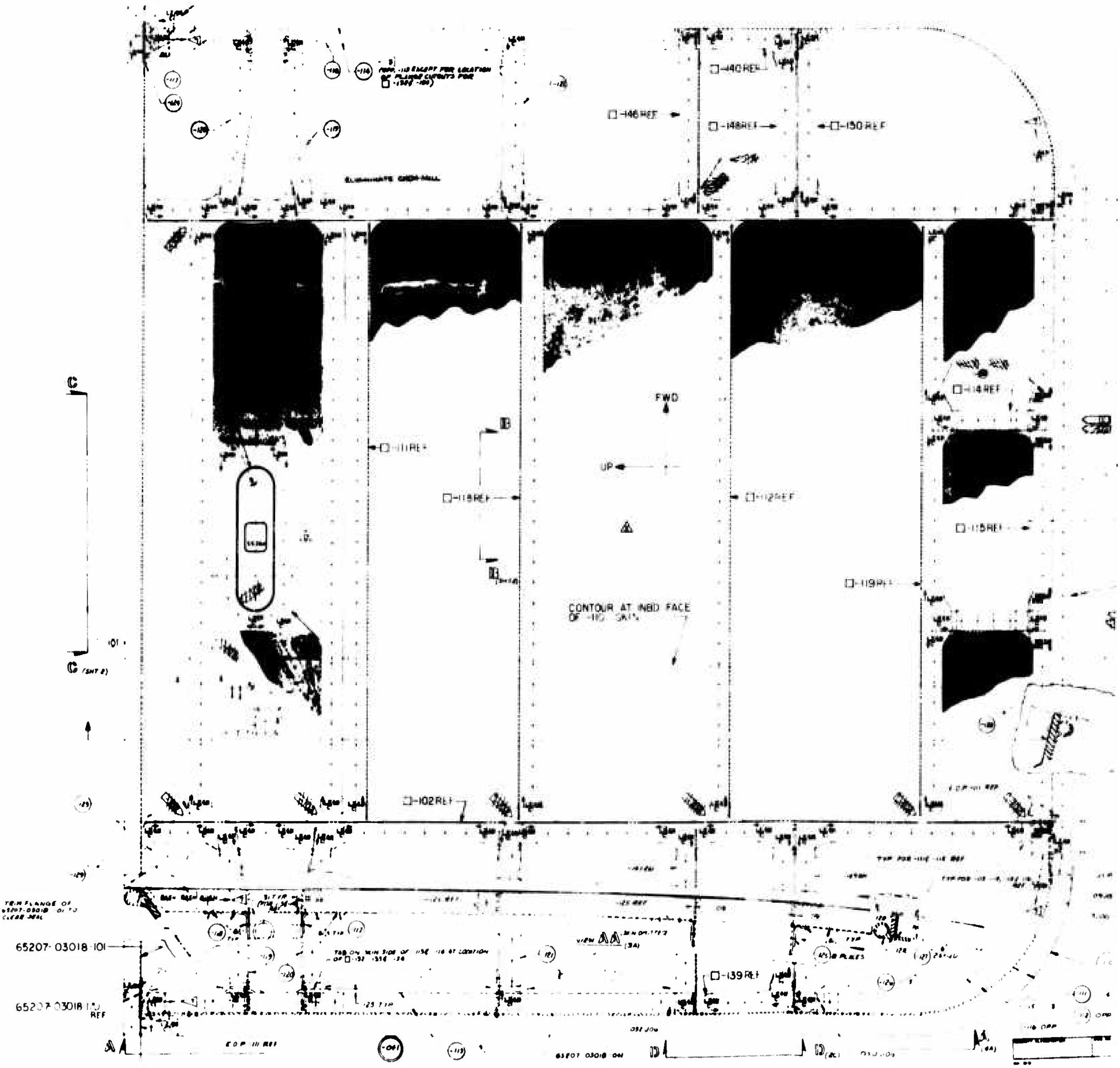

Figure 10. Redesigned Lower Personnel Door (Drawing 38008-20002, Sheet 1).

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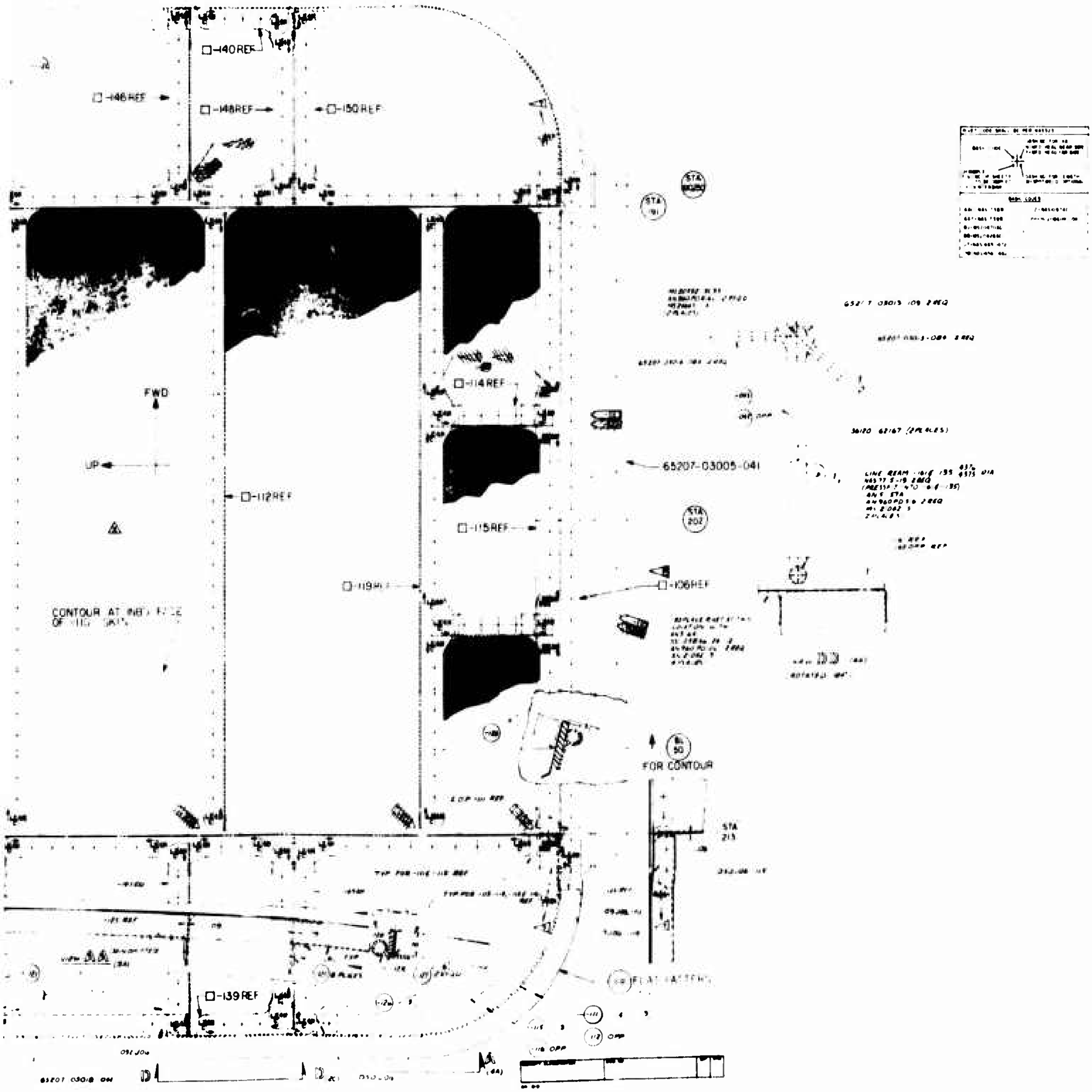

rsonnel Door
2, Sheet 1).

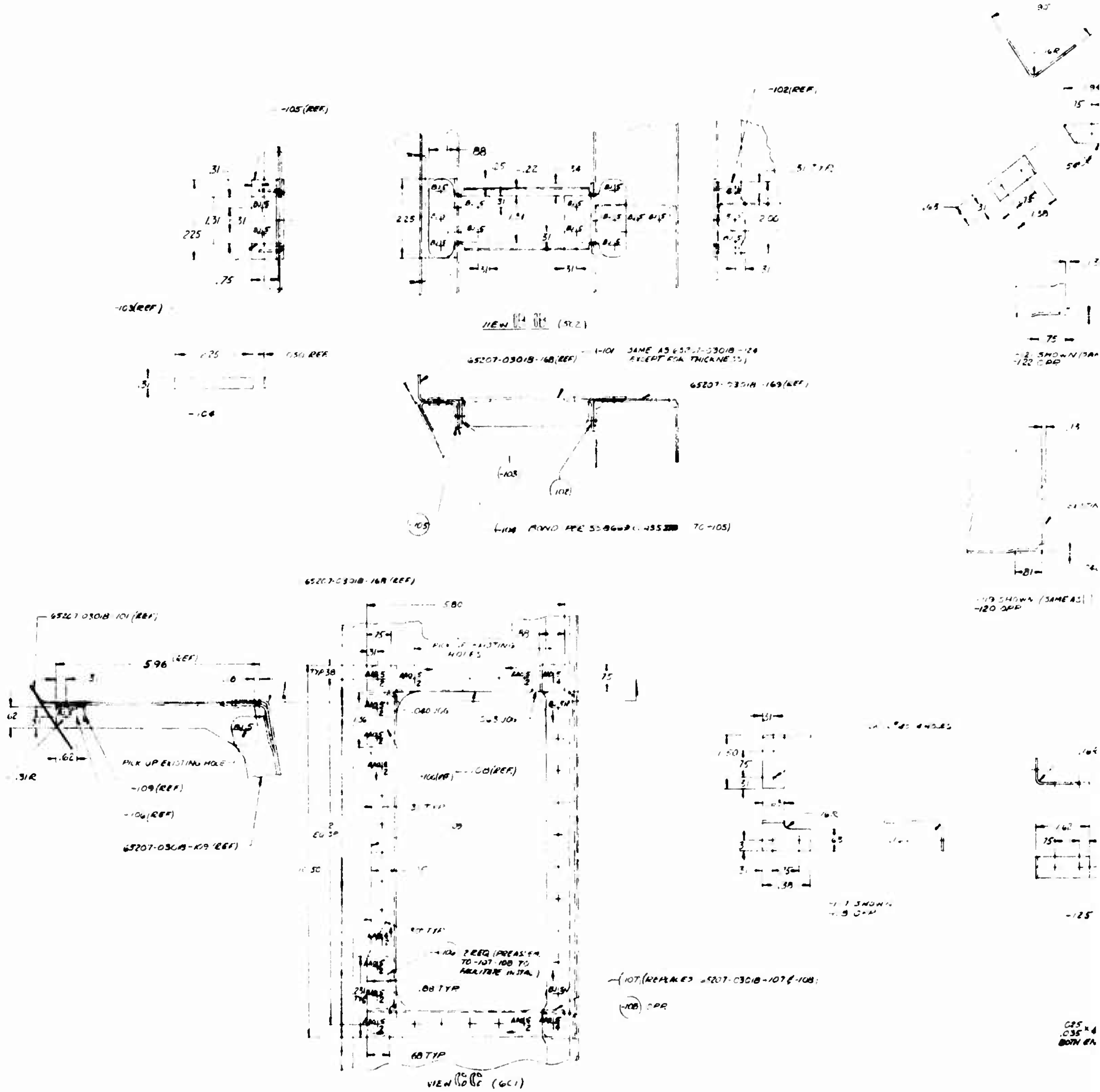

Figure 11. Redesigned Lower Personnel Door. (Drawing 38008-20002, Sheet 2).

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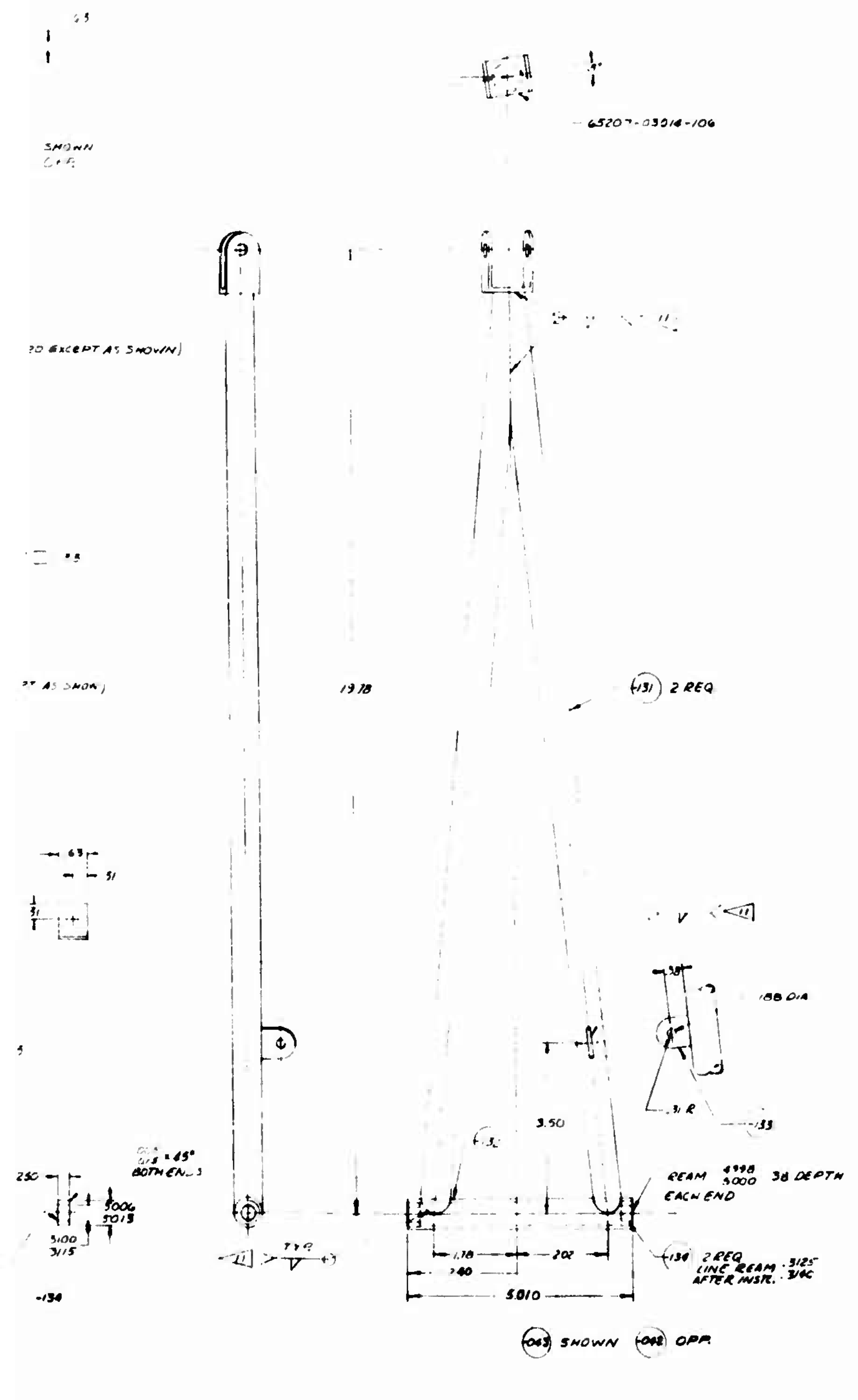
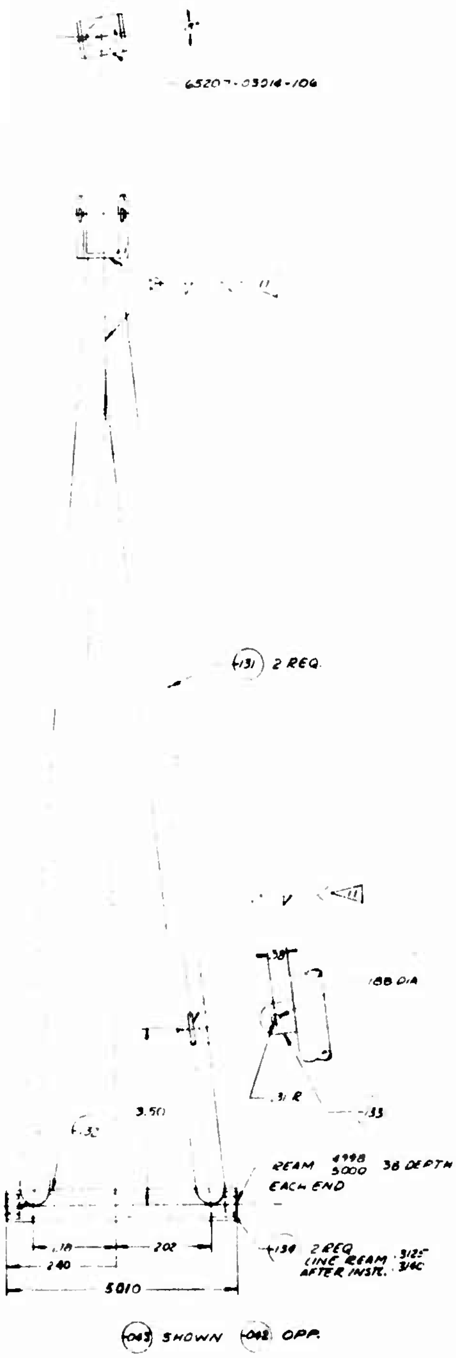


NOTES

1. AFTER FORMING, HT PER MIL-H-6088 TO T6
2. -110 SKIN IS SAME AS 65207-03011-103 EXCEPT THAT .060 IS MAINTAINED AROUND ENTIRE EDGE.
3. OBTAIN CONTOUR & TRUE LENGTHS FROM MYLAR
4. [illegible] -102 MAY BE MADE BY SPLICING REWORKED [illegible] -155 & -156. SPLICE ANGLE TO BE [illegible] 7075 T6. PICKING UP TWO RIVETS EITHER SIDE OF SPLICE
5. BOND WITH RTV-102 PER 55B640 TO -113 -114 -115 [illegible] AS SHOWN. TRANSITION AT TERMINATION OF CHANNELS AT TOP & BOTTOM EDGES OF DOOR & BOND AS NOTED IN RESPECTIVE CROSS SECTIONAL VIEWS. SEAL TO BE ONE LENGTH SPLICED AT CENTER OF TOP OF DOOR
6. BOND WITH RTV-102 PER 55B640
[illegible] EQUALLY SPACED APPROX .75 ON CENTERS
8. SYMBOL [ ] REPRESENTS DWG 65207-03[illegible]
9. THE FOLLOWING COMPONENTS OF [illegible] NOT TO BE USED -124, -125, -133, -134 [illegible] & 65207-03030-105 [illegible] FOR DETAILS.
[illegible] BOND [illegible] 65207-03019-101 TO [illegible] WITH 55B671
[illegible] WELD PER MIL-W-8604 USING GG-R [illegible] FILLER WIRE. AFTER WELD HT PER [illegible] 6061-T6
[illegible] BEFORE FINISH MACH HT TO 125,000 PSI ROCKWELL C26-33 PER MIL-H-6875
3. -130 PAN TO BE SAME AS 65207-03[illegible] THAT .13 [illegible] CORNER RADII TO BE [illegible]

QTY	CODE IDENT NO	PART NUMBER	NOMENCLATURE	SPECIFICATION
4		55103[illegible]-24-[illegible]	RADIUS BLOCK	
4		AN960PD516	WASHER	
4		AN960PD416L	WASHER	
[illegible]		AN960PD10L	WASHER	
2		MS21042-5	NUT	
4		MS21042-3	NUT	
4		AN5-51A	BOLT	
4		AN3-6A	BOLT	
[illegible]		MS24665-151	COTTER PIN	
[illegible]		MS20392-3C35	PIN	
4		NAS77-5-19	BUSHING	
AR		RTV-102	ADHESIVE	
AR		NAS1738B[illegible]	RIVET	
AR		NAS1738B[illegible]	RIVET	
AR		NAS1097AD4	RIVET	
AR		MS20426AD5	RIVET	
AR		MS20470AD5	RIVET	
AR		MS20470AD4	RIVET	
[illegible]		[illegible]	SPRING	
[illegible]		65207-03014-[illegible]	CABLE	
[illegible]		65207-[illegible]	CABLE	
[illegible]		65207-03014-10[illegible]	CABLE STOP	
[illegible]		65207-03014-106	CHANNEL	
[illegible]		65207-03019-101	SUPPORT	
1		-134	BUSHING	MIL-S-6758
1		-133	LUG	QQ-A-[illegible]
1		-132	TUBE	WW-[illegible]
2		-131	TUBE	WW-[illegible]
[illegible]		[illegible]-20002-130	LATCH PAN	
[illegible]		-129	FILLER	AMS 3[illegible]
[illegible]		-128	FILLER	AMS 3195
[illegible]		-127	FILLER	AMS 3195
2		-126	SEAL	MAKE FROM [illegible]
[illegible]		-125	CLIP	QQ-A-250/
3		-124	CLIP	QQ-A-250/
[illegible]		-123	CLIP	QQ-A-250/
[illegible]		-122	ANGLE	QQ-A-[illegible]
[illegible]		-121	ANGLE	QQ-A-250/
[illegible]		-120	SUPPORT	QQ-A-250/
[illegible]		-119	SUPPORT	QQ-A-250/
[illegible]		-118	ANGLE	QQ-A-250/
[illegible]		-117	ANGLE	QQ-A-250/
[illegible]		-116	CHANNEL	QQ-A-250/
[illegible]		-115	CHANNEL	QQ-A-250/
[illegible]		-114	CHANNEL	QQ-A-250/
[illegible]		-113	CHANNEL	QQ-A-250/
[illegible]		-112	ANGLE	QQ-A-250/
[illegible]		-111	ANGLE	QQ-A-250
[illegible]		-110	SKIN	QQ-A-250/
[illegible]		-109	ANGLE	QQ-A-250/
[illegible]		-108	ANGLE	QQ-A-250/
[illegible]		-107	ANGLE	QQ-A-250/
[illegible]		-106	ANGLE	QQ-A-200/
2		-105	ANGLE	QQ-A-200/
[illegible]		-104	FILLER	QQ-A-250/
[illegible]		-103	ANGLE	QQ-A-200/
[illegible]		-102	ANGLE	QQ-A-200/
1		38008-20002-101	SUPPORT CHANNEL	QQ-A-250/
1		38008-20002-043	WELD ASSEM	
1		38008-20002-042	WELD ASSEM	
[illegible]		65207-03018-04[illegible]	DOOR ASSY LOWER	
[illegible]		38008-20002-[illegible]	DOOR ASSY LOWER	

LIST OF MATERIAL AND PARTS



NOTES

1. AFTER FORMING, H.T. PER MIL-H-6088 TO T6.
2. -110 SKIN IS SAME AS 65207-03011-103 EXCEPT THAT .060 IS MAINTAINED AROUND ENTIRE EDGE.
3. OBTAIN CONTOUR & TRUE LENGTHS FROM MYLAR
4. [illegible] MAY BE MADE BY SPLICING RENEWED [illegible] SPLICE ANGLE TO BE [illegible] 7075 [illegible] RIVETING IN TWO RIVETS EITHER SIDE OF SPLICE
5. BOND WITH RTV102 PER [illegible] TO -113 [illegible] -155 [illegible] AS SHOWN. TRANSITION AT TERMINATION OF CHANNELS AT TOP & BOTTOM EDGES OF DOOR & BOND AS NOTED IN RESPECTIVE CROSS SECTIONAL VIEWS. SEAL TO BE ONE LENGTH SPLICED AT CENTER OF TOP OF DOOR
6. BOND WITH RTV102 PER 558640
7. EQUALLY SPACED APPROX .75 ON CENTERS
8. SYMBOL [ ] REPRESENTS DWG 65207-03018
9. THE FOLLOWING COMPONENTS OF 65207-03018 ARE NOT TO BE USED: -124, -125, -133, -134 [illegible] 65207-03050-105 [illegible] SEE VIEWS AA, B-B & [illegible] FOR DETAILS
10. BOND [illegible] 65207-03018-101 TO -110 IN ACCORD WITH 558671
11. WELD PER MIL-W-8604 USING QQ-R-566 CL 4043 FILLER WIRE. AFTER WELD H.T. PER MIL-H-6088 TO 6061-T6
12. BEFORE FINISH MACH. H.T. TO 125,000 PSI MIN. T.S. ROCKWELL C26-33 PER MIL-H-6875
13. -130 PAN TO BE SAME AS 65207-03018-124 EXCEPT THAT .13 MAX CORNER RADII TO BE USED

QTY	PART NUMBER	NOMENCLATURE	SPECIFICATION	STOCK SIZE	MATERIAL OR NOTE
[illegible]	[illegible]	[illegible] BLOCK			
4	AN960[illegible]	WASHER			
4	AN960[illegible]	WASHER			
4	AN960[illegible]	WASHER			
[illegible]	MS21042-[illegible]	NUT			
[illegible]	MS21042-[illegible]	NUT			
4	AN3-[illegible]	BOLT			
4	AN3-6A	BOLT			
[illegible]	MS[illegible]	COTTER PIN			
[illegible]	[illegible]	PIN			
4	NAS77-3-[illegible]	BUSHING			
[illegible]	[illegible]	RETAINER			
AR	NAS[illegible]	RIVET			
AR	NAS[illegible]	RIVET			
AR	NAS[illegible]	RIVET			
AR	MS[illegible]	RIVET			
AR	MS[illegible]	RIVET			
AR	MS[illegible]	RIVET			
[illegible]	[illegible]	SPRING			
[illegible]	65207-[illegible]	SEAL			
[illegible]	65207-[illegible]	SEAL			
[illegible]	65207-[illegible]	[illegible]			
[illegible]	65207-03014-[illegible]	CHANNEL			
[illegible]	65207-03018-[illegible]	SUPPORT			
[illegible]	-[illegible]	BUSHING	MIL-S-6758	.625 DIA	[illegible] CONDITION D
[illegible]	-[illegible]	[illegible]	QQ-A-250/11	.050	6061-T6
[illegible]	-[illegible]	TUBE	WW-T-700/6	[illegible]	6061-T6
2	-[illegible]	TUBE	WW-T-700/6	[illegible]	6061-T6
2	[illegible]	[illegible] PAN			
[illegible]	-[illegible]	FILLER	AMS 3195	[illegible]	SPONGE RUBBER
[illegible]	-[illegible]	FILLER	AMS 3195	[illegible]	SPONGE RUBBER
[illegible]	-[illegible]	FILLER	AMS 3195	[illegible]	SPONGE RUBBER
[illegible]	-[illegible]	[illegible]	MAKE FROM 65207-0301[illegible]	[illegible]	[illegible]
[illegible]	-[illegible]	CLIP	QQ-A-250/[illegible]	.050	7075-T6
[illegible]	-[illegible]	CLIP	QQ-A-250/[illegible]	.050	7075-T6
[illegible]	-[illegible]	CLIP	QQ-A-250/[illegible]	.050	7075-T6
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	[illegible]	[illegible]
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	[illegible]	[illegible]
[illegible]	-[illegible]	[illegible]	QQ-A-250/[illegible]	[illegible]	[illegible]
[illegible]	-[illegible]	[illegible]	QQ-A-250/[illegible]	[illegible]	[illegible]
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	[illegible]	[illegible]
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	[illegible]	[illegible]
[illegible]	-[illegible]	CHANNEL	QQ-A-250/[illegible]	[illegible]	7075-T6
[illegible]	-[illegible]	CHANNEL	QQ-A-250/[illegible]	[illegible]	7075-T6
[illegible]	-[illegible]	CHANNEL	QQ-A-250/[illegible]	.080	7075-T6
[illegible]	-[illegible]	CHANNEL	QQ-A-250/[illegible]	.080	7075-T6
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	.080	7075-T6
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	.080	7075-T6
[illegible]	-110	SKIN	QQ-A-250/[illegible]	[illegible]	7075-T6
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	[illegible]	7075-T6
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	[illegible]	7075-T6
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	[illegible]	7075-T6
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	[illegible]	7075-T6
2	-[illegible]	ANGLE	QQ-A-250/[illegible]	[illegible]	7075-T6
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	.050	7075-T6
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	MAKE FROM AND [illegible]	
[illegible]	-[illegible]	ANGLE	QQ-A-250/[illegible]	MAKE FROM AND [illegible]	
[illegible]	[illegible]	SUPPORT CHANNEL	[illegible]	.050	[illegible]
[illegible]	[illegible] 20002-[illegible]	WELD ASSEM			
[illegible]	[illegible] 20002-[illegible]	WELD ASSEM			
[illegible]	65207-03018-[illegible]	DOOR ASSY LOWER			
[illegible]	58008-20002-[illegible]	DOOR ASSY LOWER			

LIST OF MATERIAL AND PARTS LIST

TEST

Main Rotor Pylon Hinged Cover

With the main rotor pylon hinged cover as originally designed (Figure 12), a slight amount of abuse was enough to cause a series of malfunctions and damage modes (Table XIV). All of the damage was caused by the first five abusive slams (open and close) on the hinged cover test schedule (Table IX). The hinged cover was then put onto the test fixture for the vibratory load (Figure 13). The vibration phase was halted after 90 hours running when it was found that the localized loading was cracking the hinged cover structure (Figures 14 and 15) which was not a realistic field mode.

Major problem areas with the original design hinged cover included the latch assembly (Figure 16) which is susceptible to jamming. The fragility of the structure showed after one abusive slam (Figures 17 and 18), which pulled rivets and buckled stringers. The aft lower corner alignment pin mounting areas are easily bent (Figure 19). The forward lower corner alignment bushings interfere with the fiber glass shell (Figure 20).

The result is that the cover misaligns so that it must be manhandled into the closed position (Figure 21).

The reworked cover (Figure 22) survived the abusive slams intact, except for the lower aft corner alignment pin mounts (Figure 23). This area on the test fixture has been found not to be typical of the aircraft, so the excessive damage incurred would not be a field damage mode (Figure 24).

The modified cover did show improved performance in resisting structural damage, misalignment, and latching difficulties. The modifications (Figure 22) include the light-colored parts shown: gussets, doublers, intercostals, and latch operating arms.

Lower Personnel Door

Testing of the lower personnel door (Figure 25) brought out the weaknesses of the original design (Table XV). The rubber seal (Figure 26) around the door inner edge prevented latching without excessive slamming. Initial installation of the door was also a problem, as it had to be shimmed at the hinge to align properly.

Although 11 of 18 field damage modes were reproduced, 2 important ones, the breaking of the support cables and the cracking of the hinge, did not develop. Upon consultation with a Sikorsky field representative at North Island, San Diego CH-53 base, it was revealed that these conditions occur from accidental in-flight door openings, which are attributed to the latching problem. The resultant "snap" openings could break the cables and hinge, possibly resulting in loss of the door.

Unlike the other two test items, the vibratory loading on the door (Figure 27) resulted in fatigue cracks similar to field modes. The original design lower personnel door developed the cracks in the skin through the support

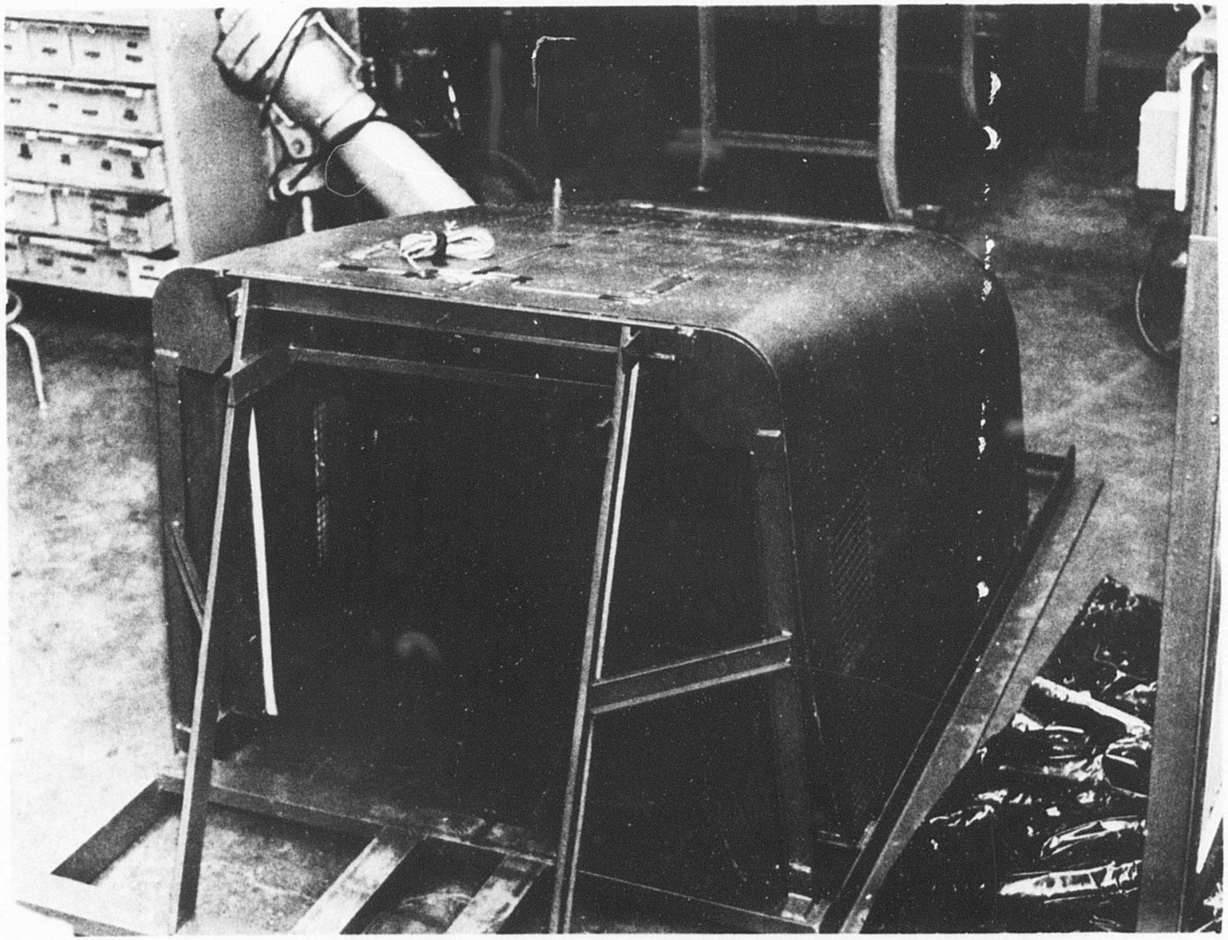


Figure 12. Main Rotor Pylon, Hinged Cover Assembly.

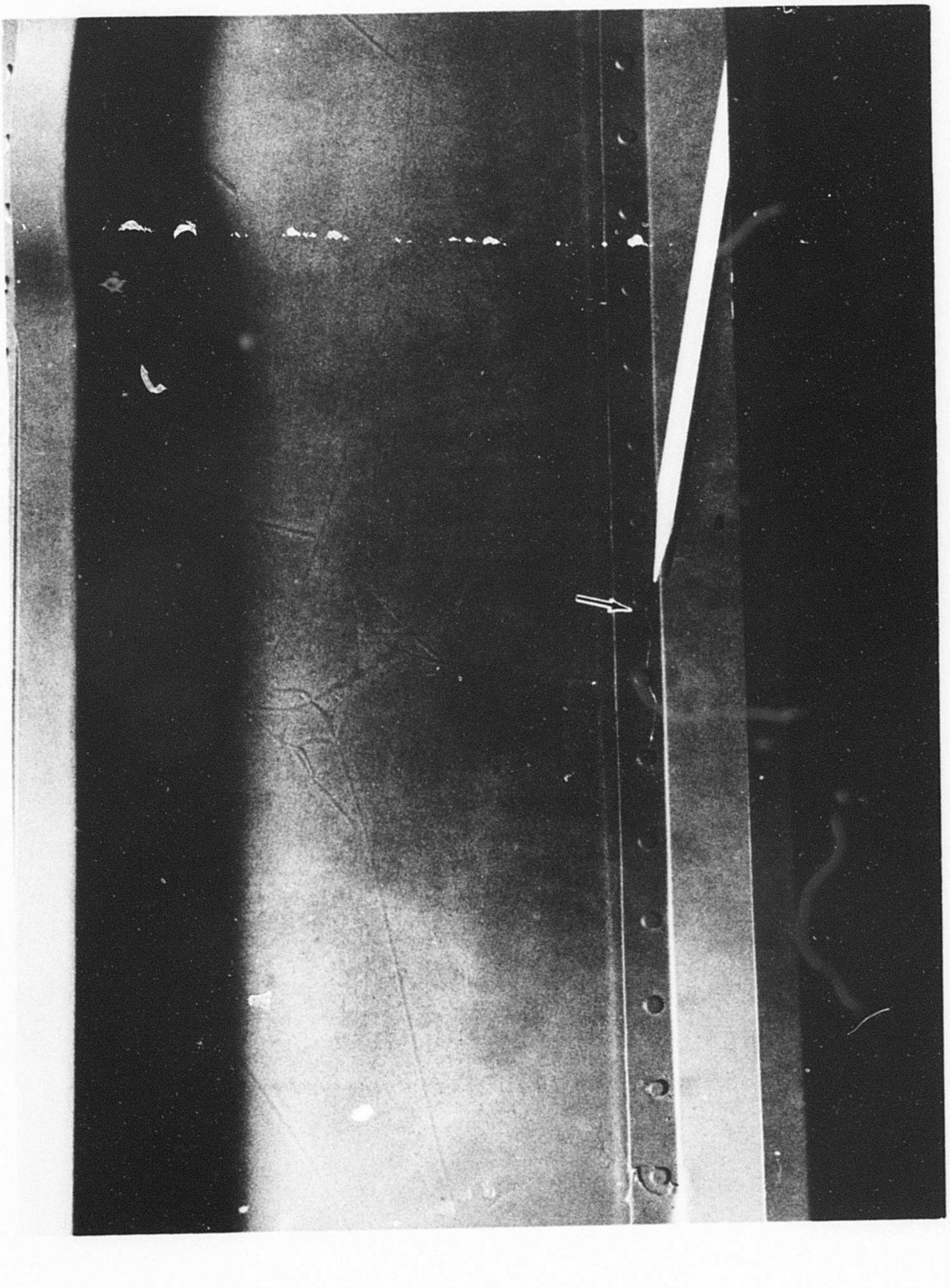


Figure 15. Main Rotor Pylon, Structural Damage, Vibratory Load.

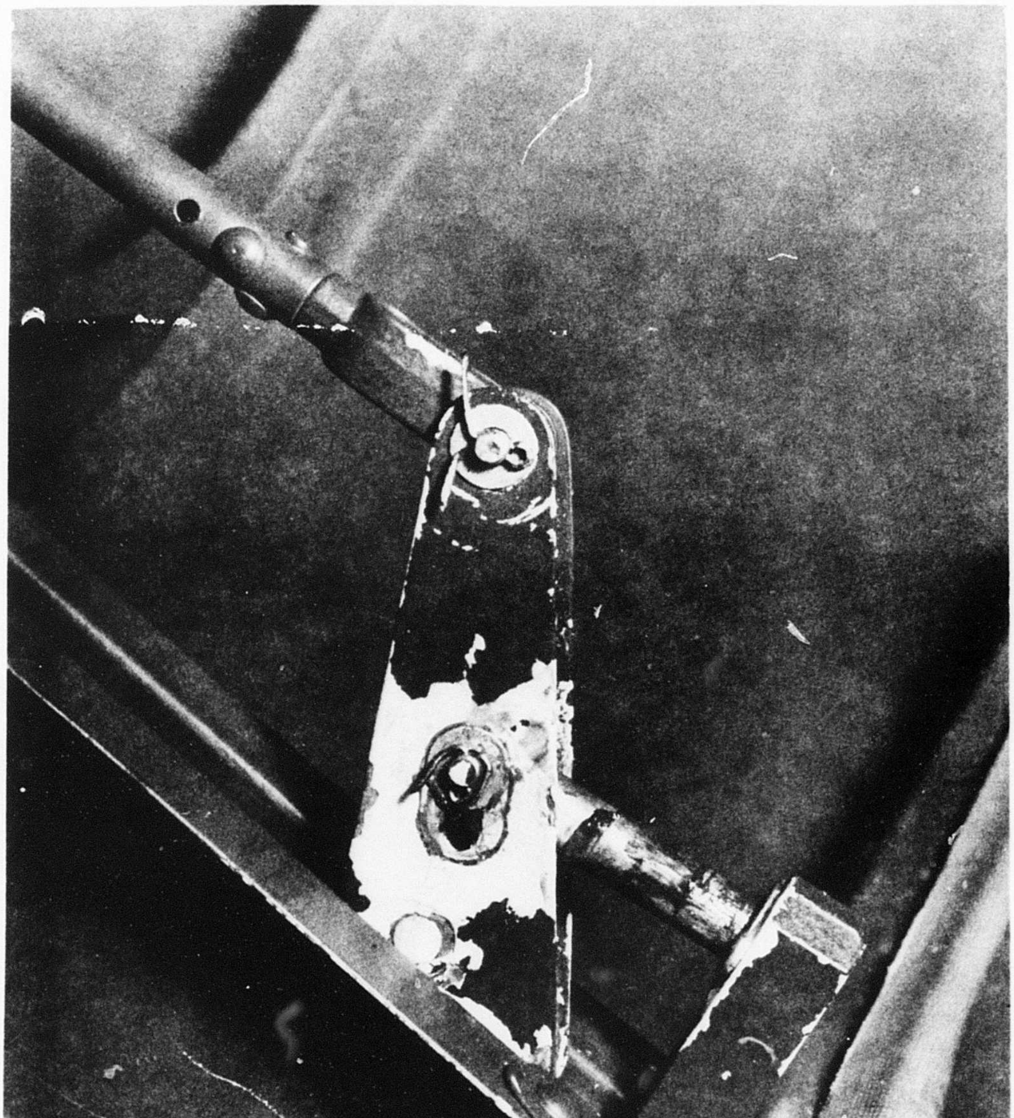


Figure 16. Main Rotor Pylon, Latch Assembly.

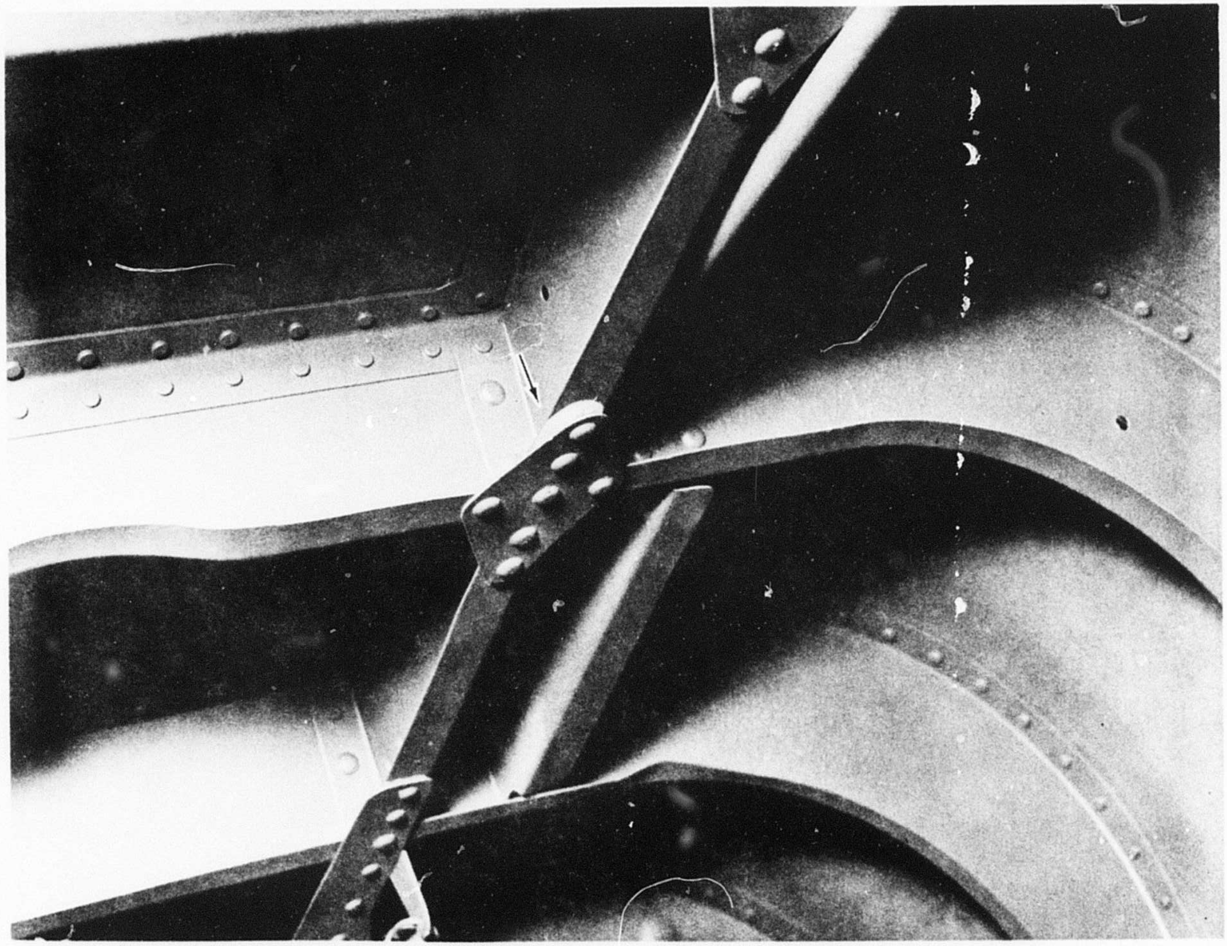


Figure 17. Main Rotor Pylon, Structural Damage, Popped Rivets.

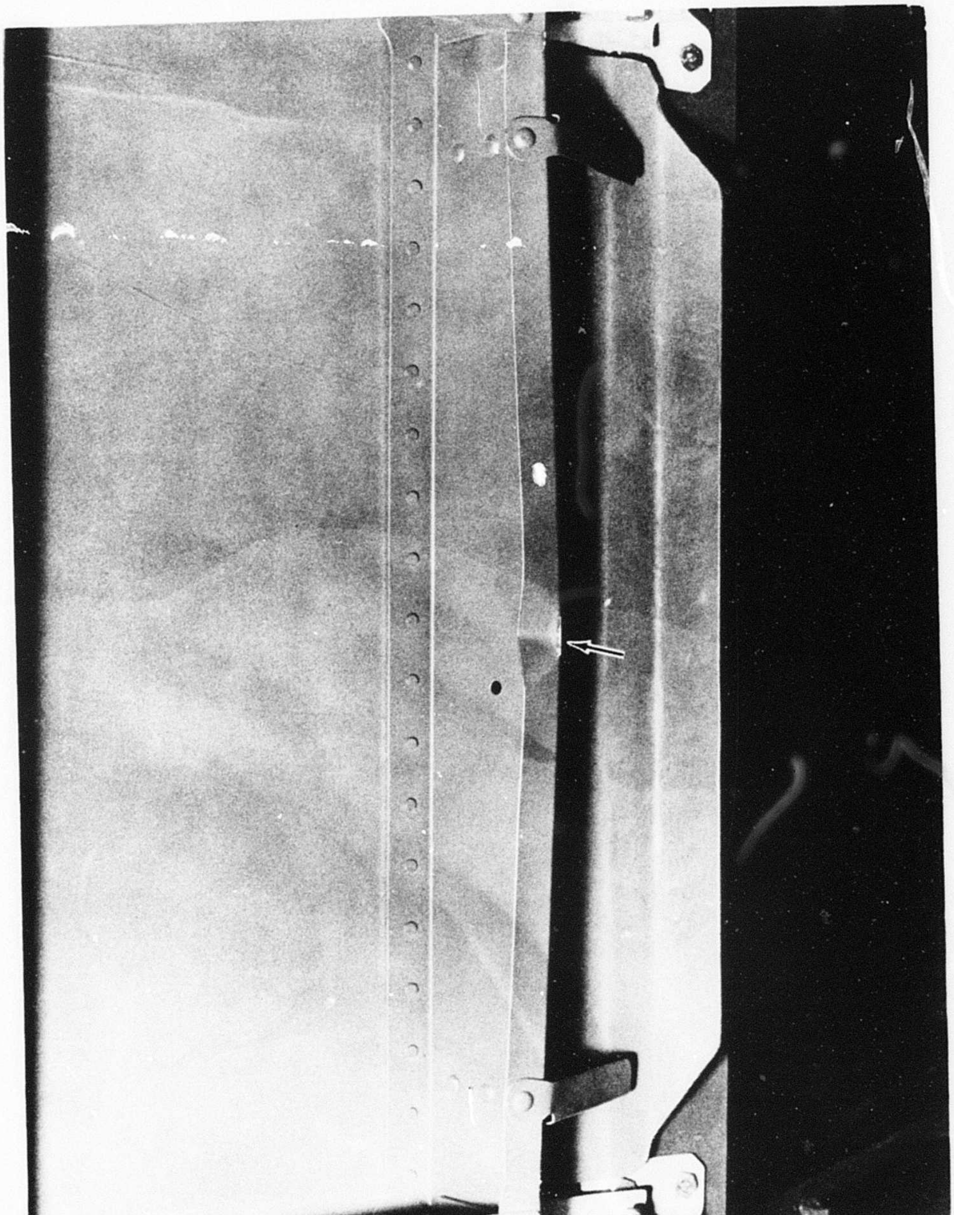


Figure 18. Main Rotor Pylon, Structural Damage, Buckled Stringer.

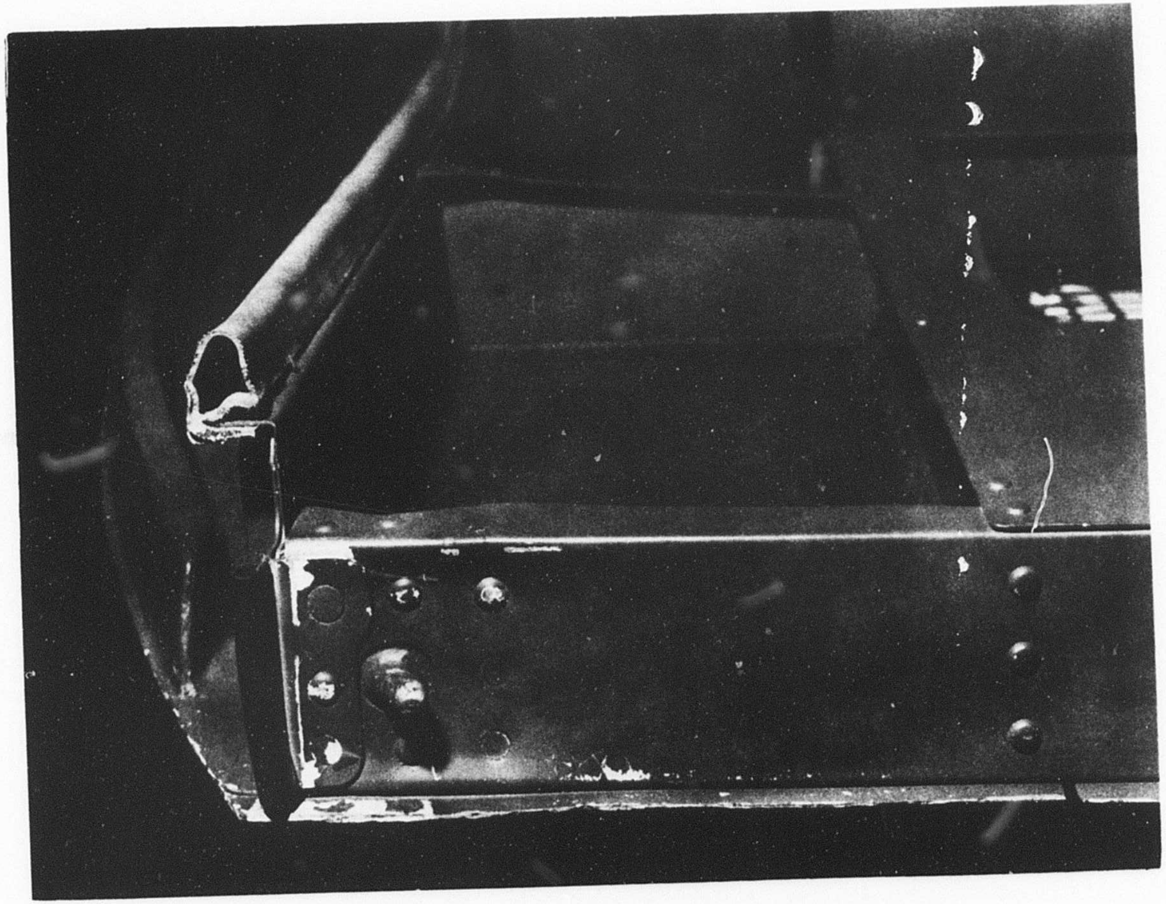


Figure 19. Main Rotor Pylon, Structural Damage (Lower Aft Corner Alignment Pin).

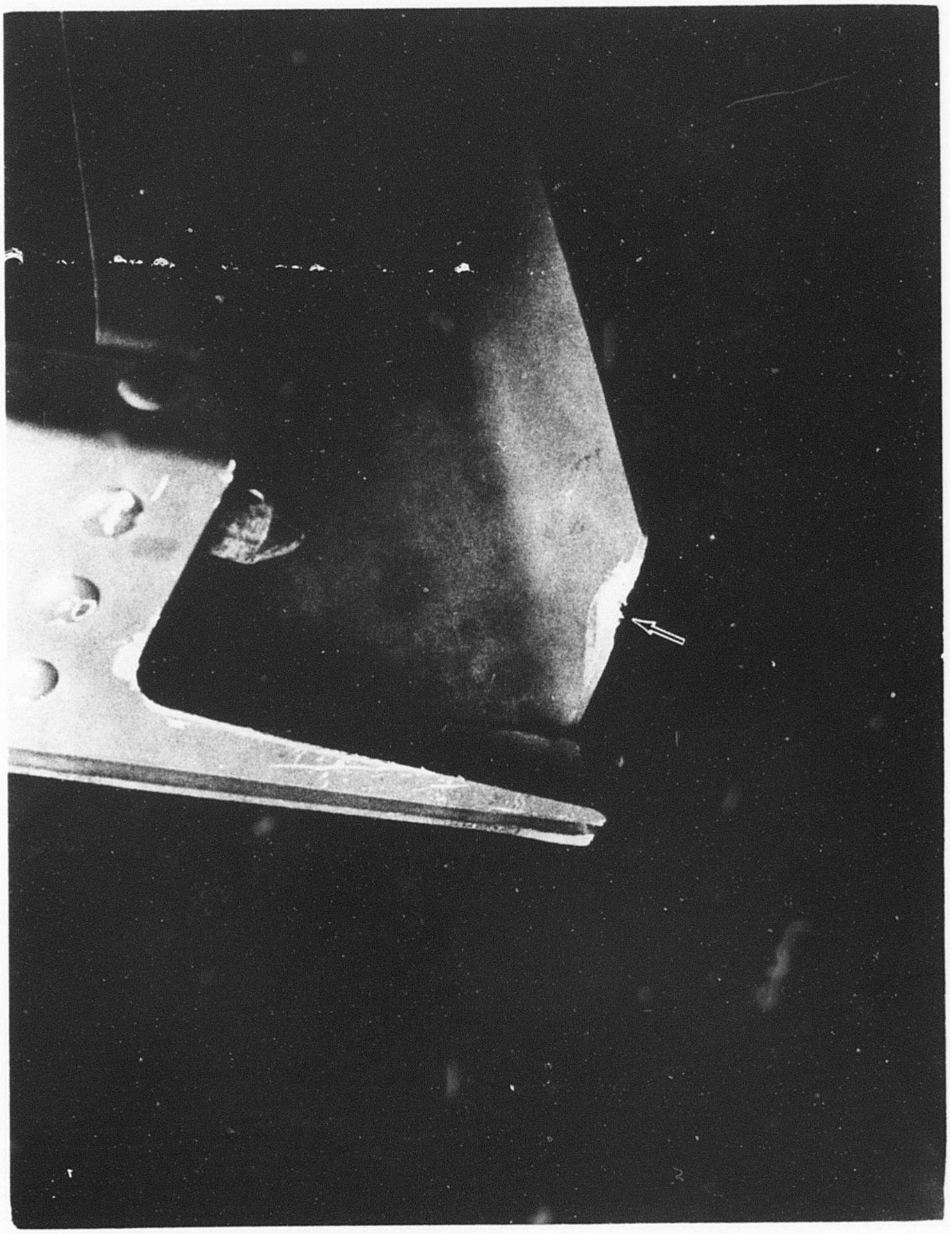


Figure 20. Main Rotor Pylon, Interference, Glass Shell.

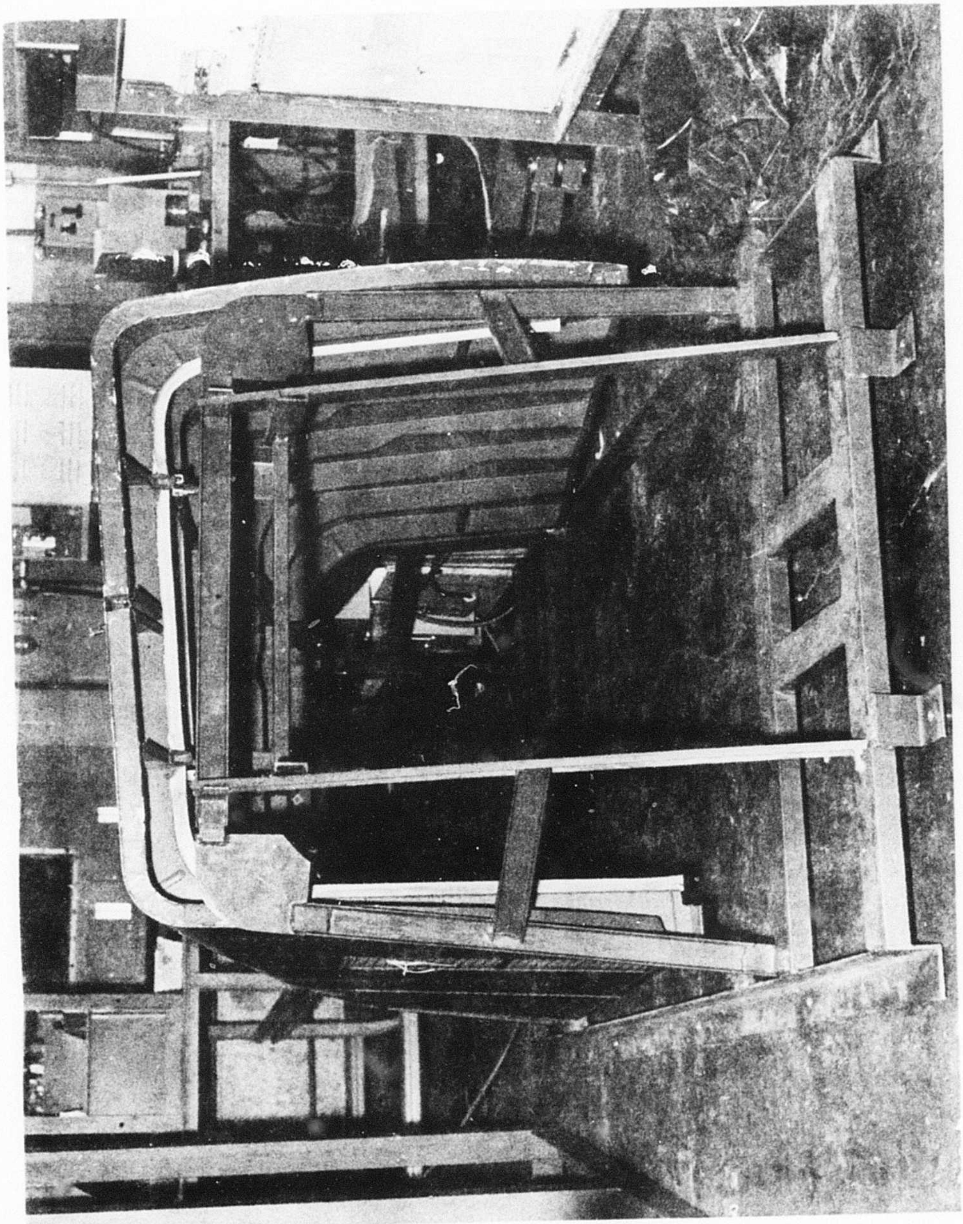


Figure 21. Main Rotor Pylon, Misalignment.

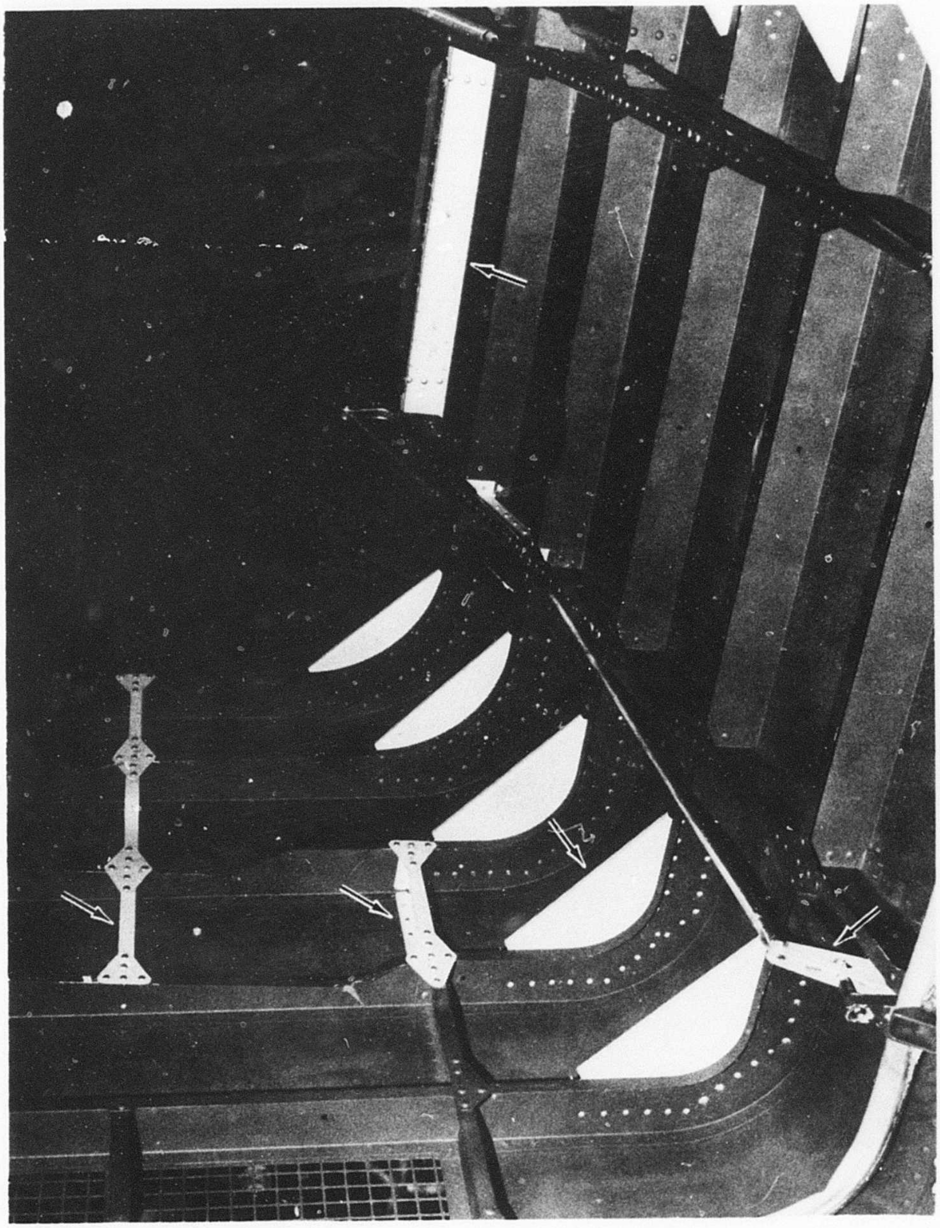


Figure 22. Main Rotor Pylon, Redesign.

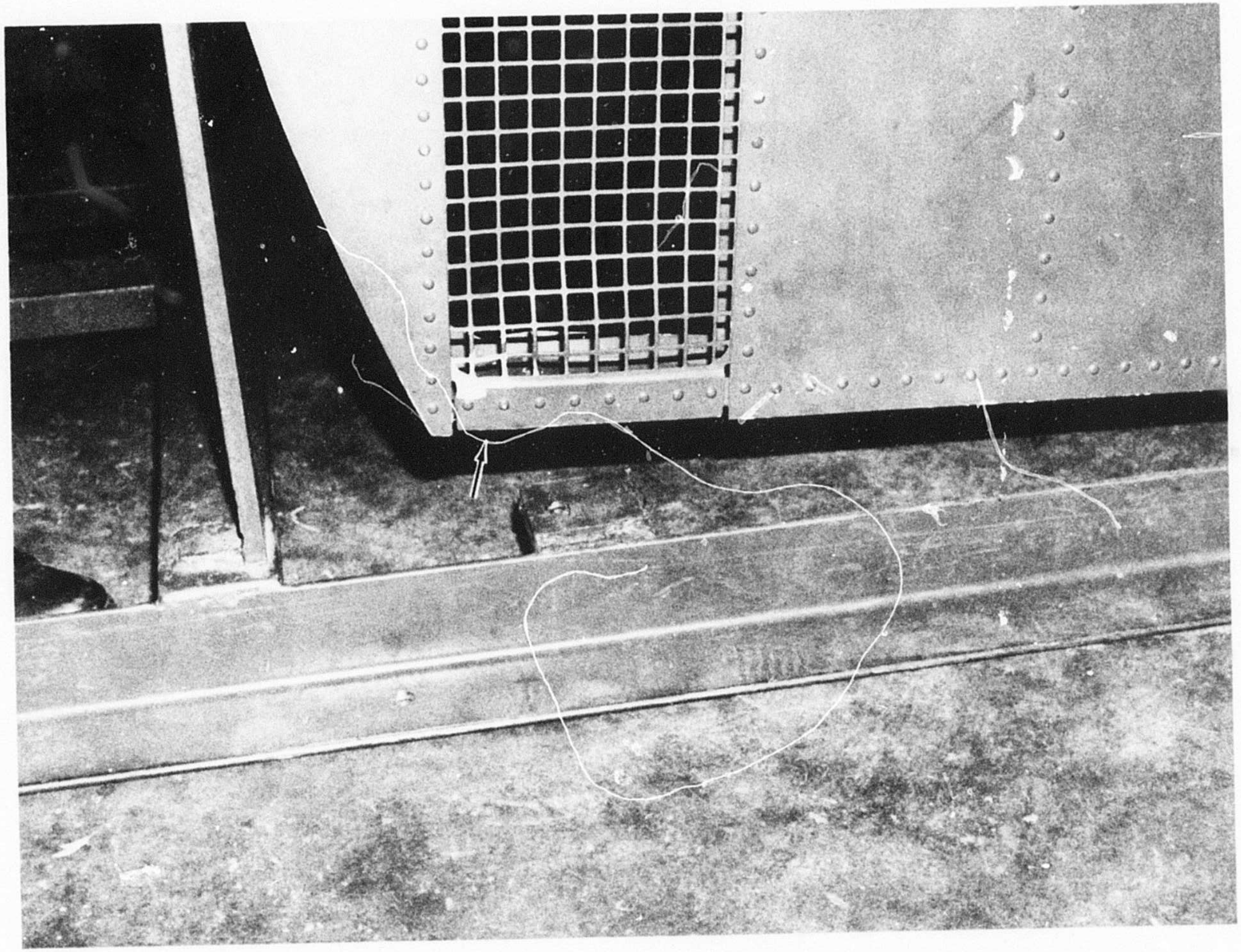


Figure 23. Main Rotor Pylon, Redesign, Alignment Pin Mount.

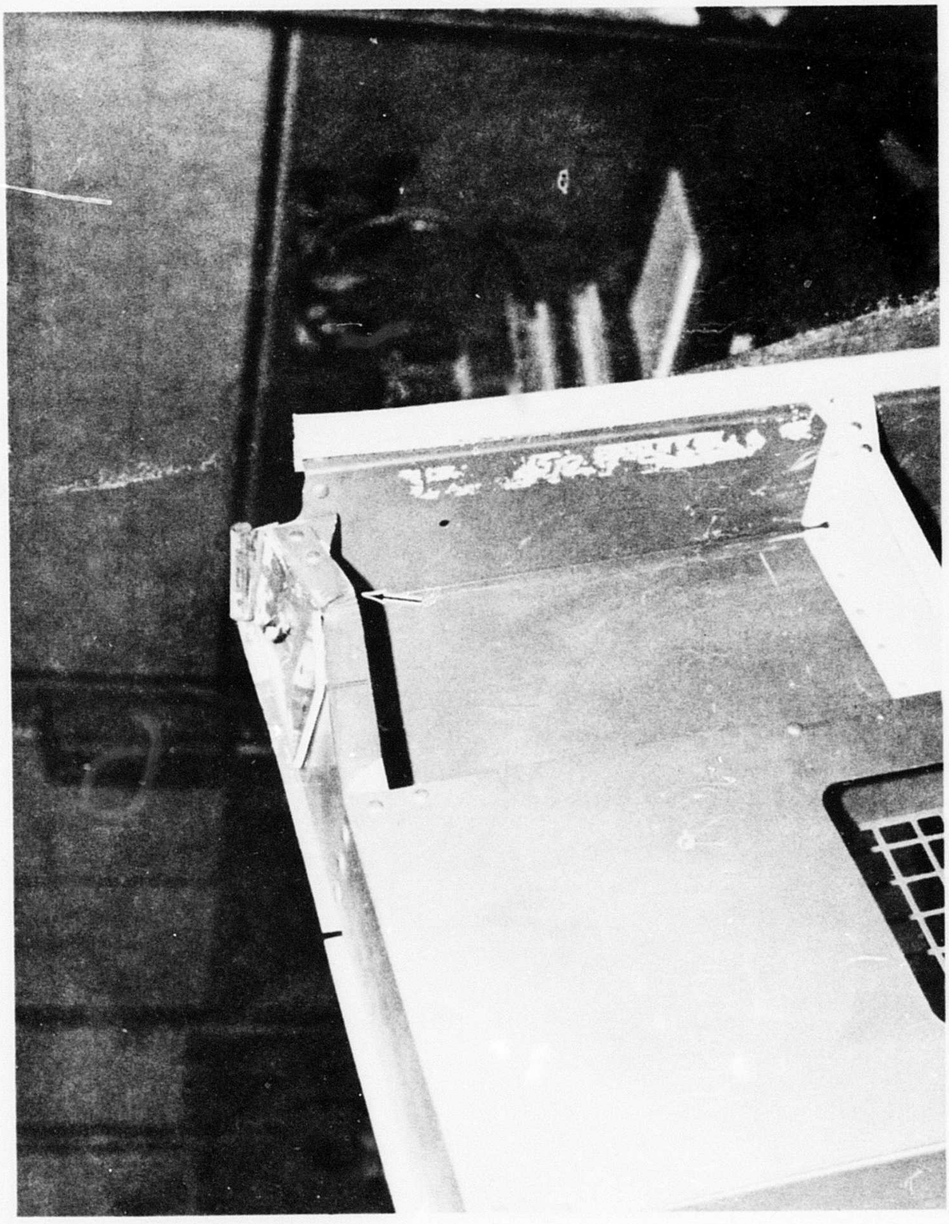


Figure 24. Main Rotor Pylon, Redesign, Alignment Pin Mount Damage.

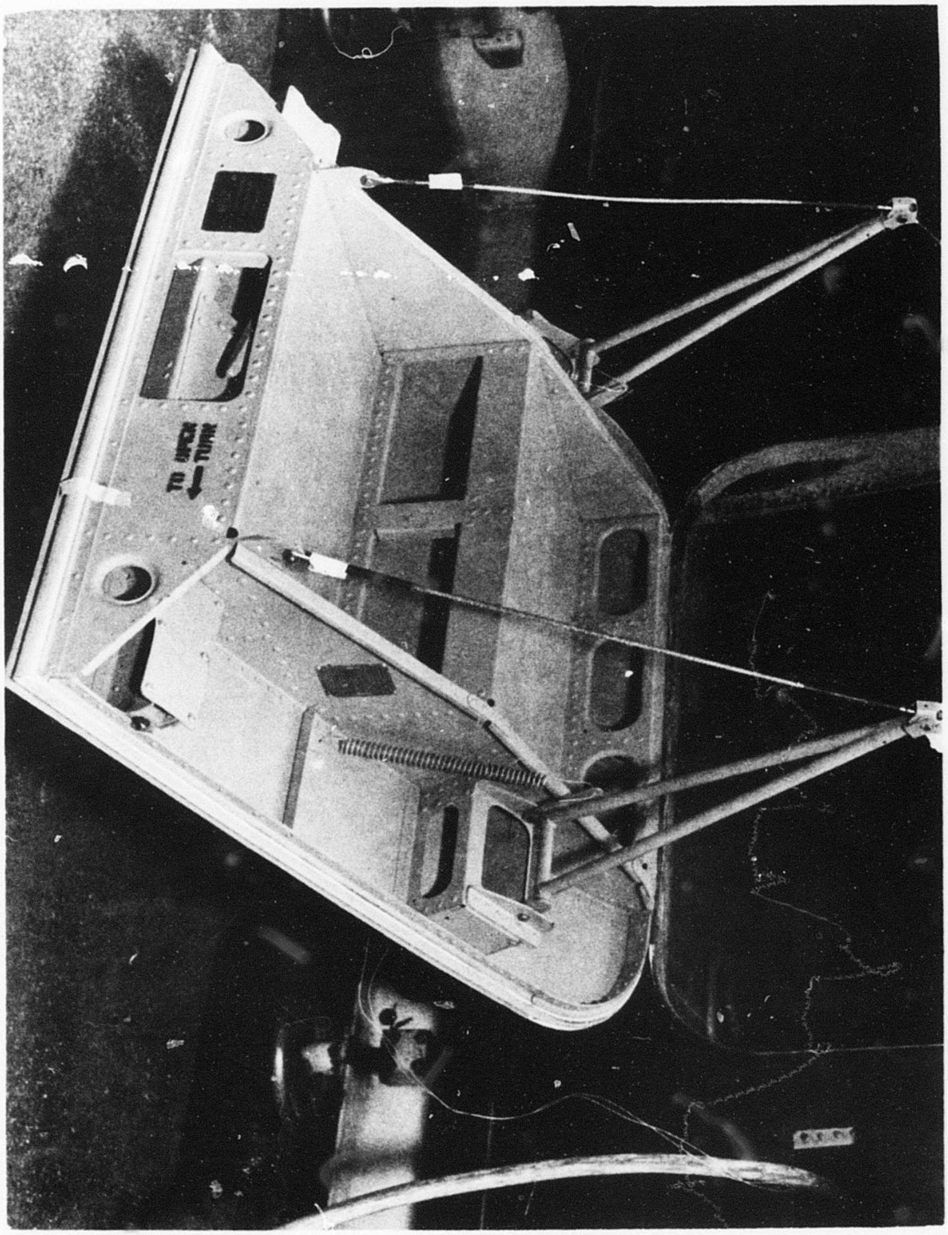


Figure 25. Lower Personnel Door.

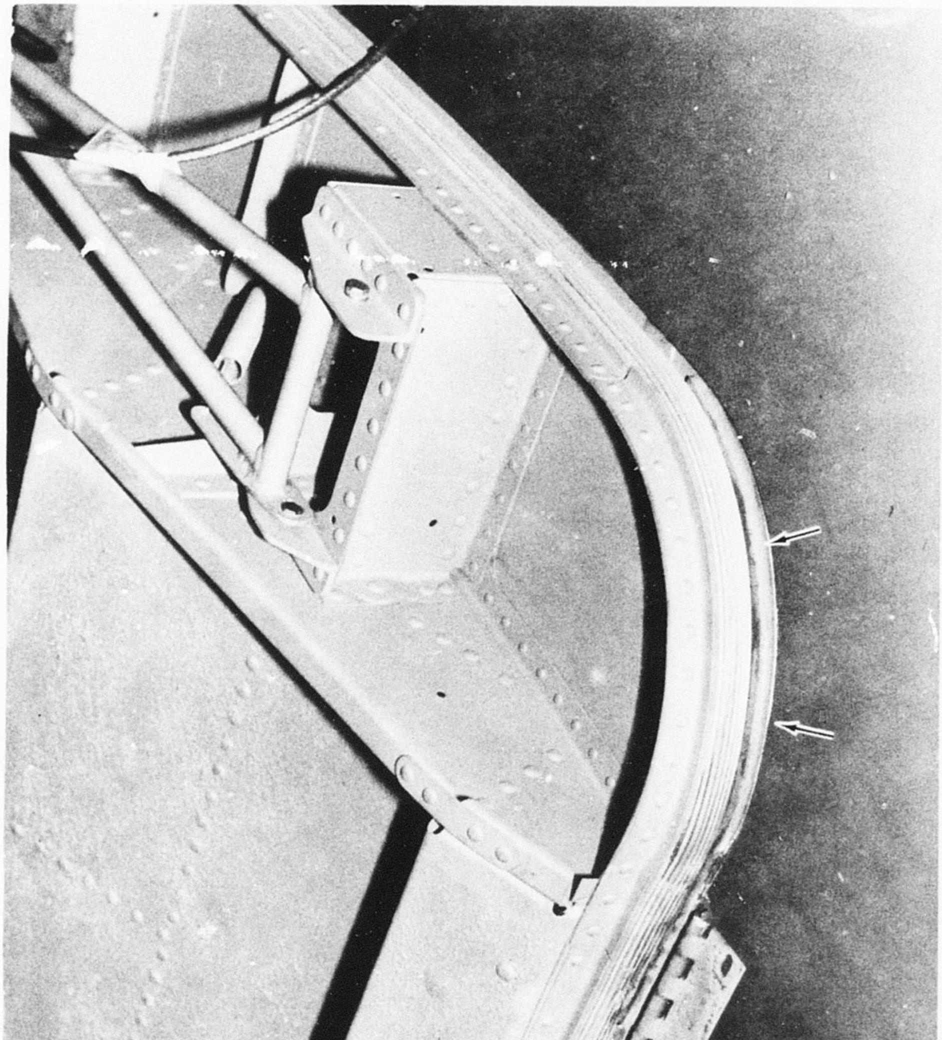


Figure 26. Lower Personnel Door, Seal, Original.

Figure 27. Lower Personnel Door, Vibratory Load Test Setup.

channels (Figures 28 and 29) at the latches in 175 hours of running. The redesigned door showed improvement. Although the same type of cracks appeared, they took longer (252 hours) and were smaller, not propagating as quickly.

Manual cycling showed that the redesigned door operated smoothly and easily without the binding seal problem (Figure 30).

Since the redesigned door is easier to latch, the possibility of a partial latching and a resulting inflight opening is minimized. A redundant catch, such as provided on automobile doors and hoods, is worth considering for such hinged structures.

Boot impacts (400 lb) on the lower step (Figure 31) produced a crack on one side beam on the original and redesigned doors (Figure 32). The boot impacts on the support cable (Figure 33) produced damage (bolt head pulling through the support) on the original door (Figure 34). The redesigned door sustained no damage on its strengthened support assembly (Figure 35).

The redesigned door was also superior in the center step riser kick test. The weighted boot was swung like a pendulum into the riser (Figure 36). The original design lasted only seven kicks (Figure 37) until fracture. The redesigned riser took 765 kicks (Figure 38).

Work Platform Assembly

The work platform assembly (Figure 39) has been failing in the field due to delamination of its fiber glass outside (weather) surface. Problems with the hinge and latches have also been reported (Table XVI).

The work platform assembly was subjected to a three-phase test: (1) vibratory load - including flight vibrations, (2) manual cycling of the platform - opening and closing, and (3) roller load - simulating men working on the platform.

The vibratory loading produced no effects (Figure 40).

Per the test schedule, the work platform assembly was also manually cycled open and closed (Figure 41).

The roller load (Figure 42) simulating men working on the platform was expected to produce the delamination field mode on the original design work platform assembly. However, this did not occur. Both the original and redesigned work platform assemblies lasted the full test without exhibiting the primary field mode of failure. The original did develop a field mode crack above the aft latch (Figure 43) after 300 hours of test when the roller load was increased to 400 pounds (2 men) from the 200-pound (1 man) loading. The crack after initial formation did not propagate further. This type of crack did not appear on the redesigned work platform assembly.

The redesigned work platform assembly is expected to eliminate the delamination problem, as it is constructed of an aluminum and honeycomb structure rather than an aluminum and fiber glass laminate on a balsa wood core.

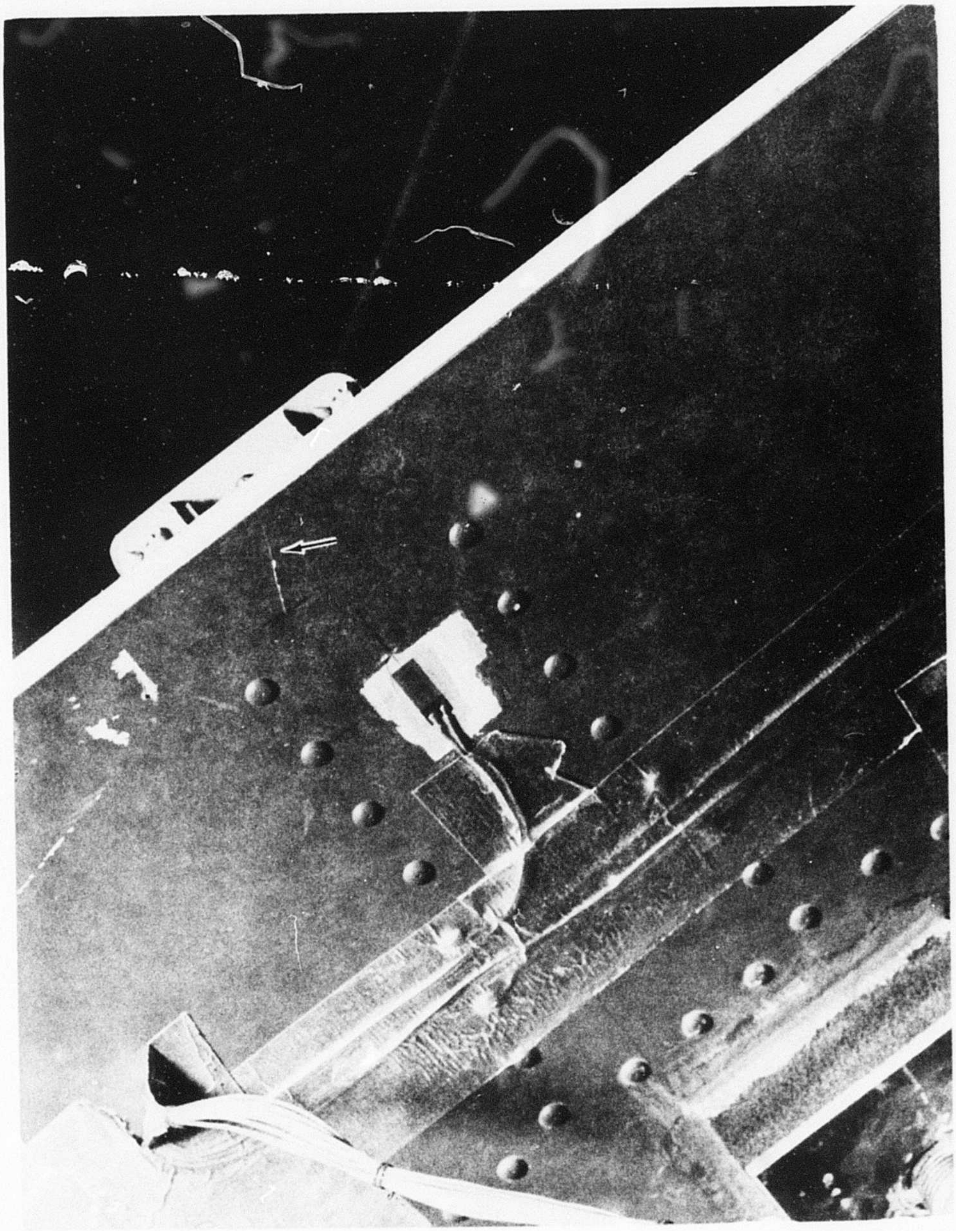


Figure 28. Lower Personnel Door, Original, Fatigue Skin Crack.

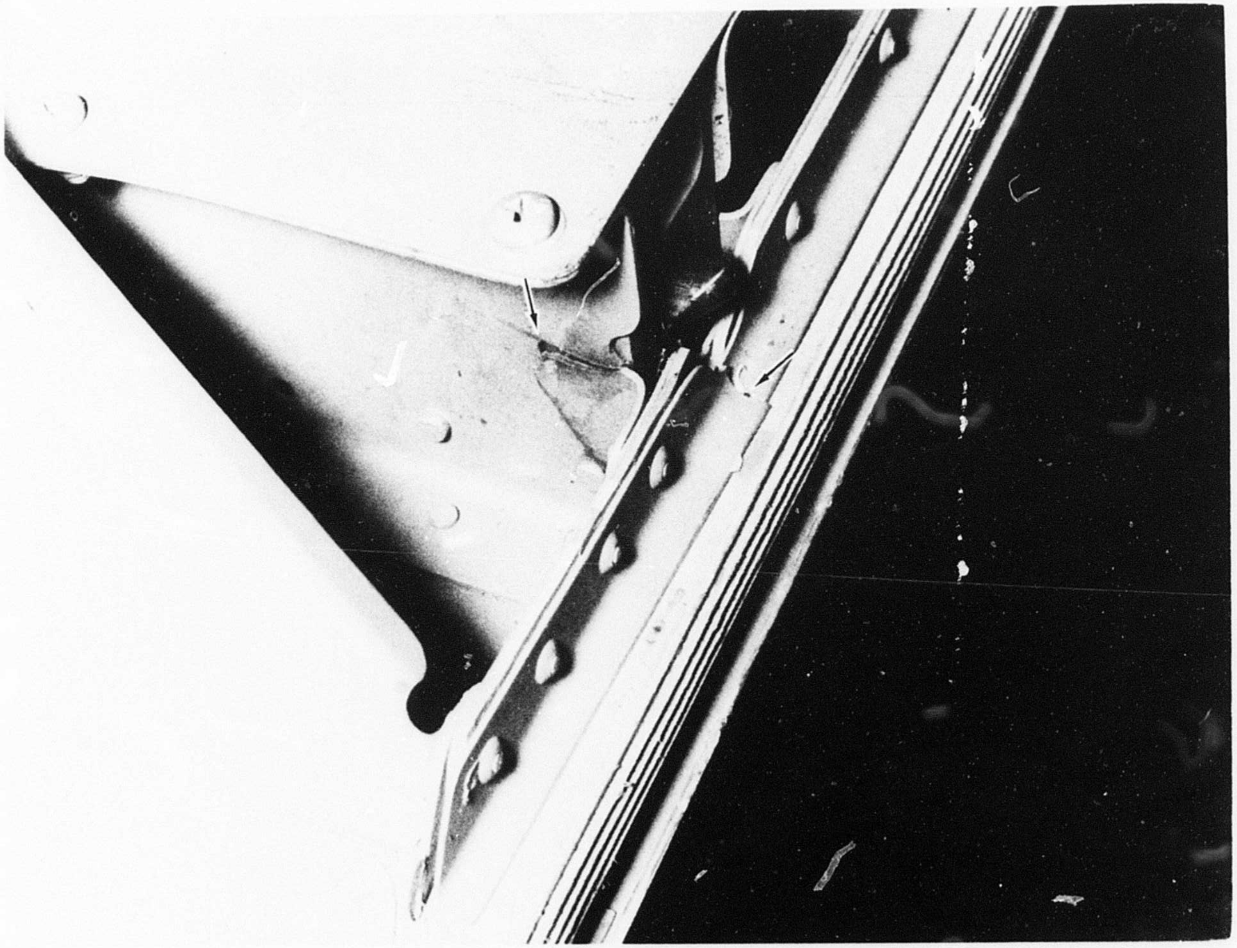


Figure 29. Lower Personnel Door, Original, Cracked Support Channels.

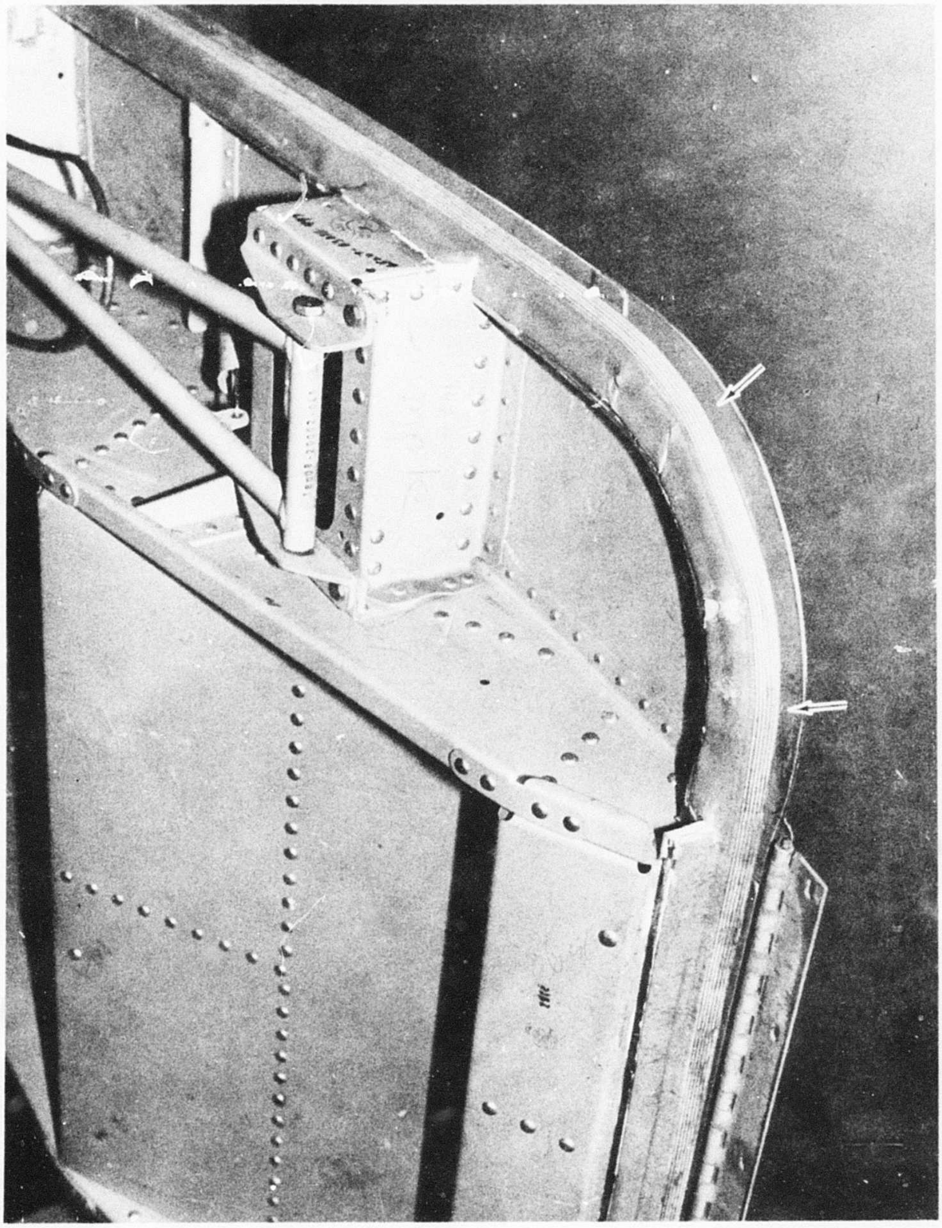


Figure 30. Lower Personnel Door, Redesign, Improved Seal.

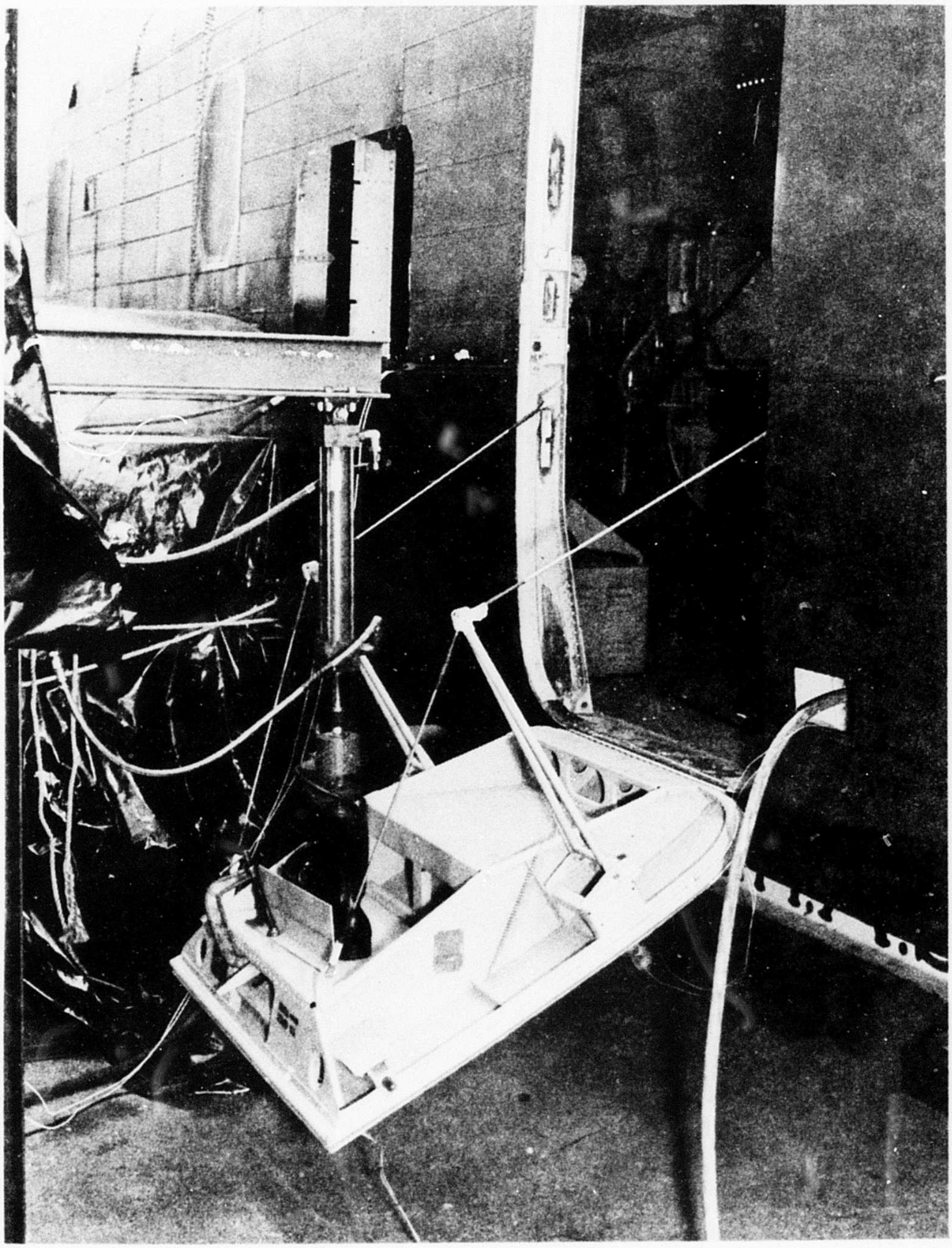


Figure 31. Lower Personnel Door, Boot Test.

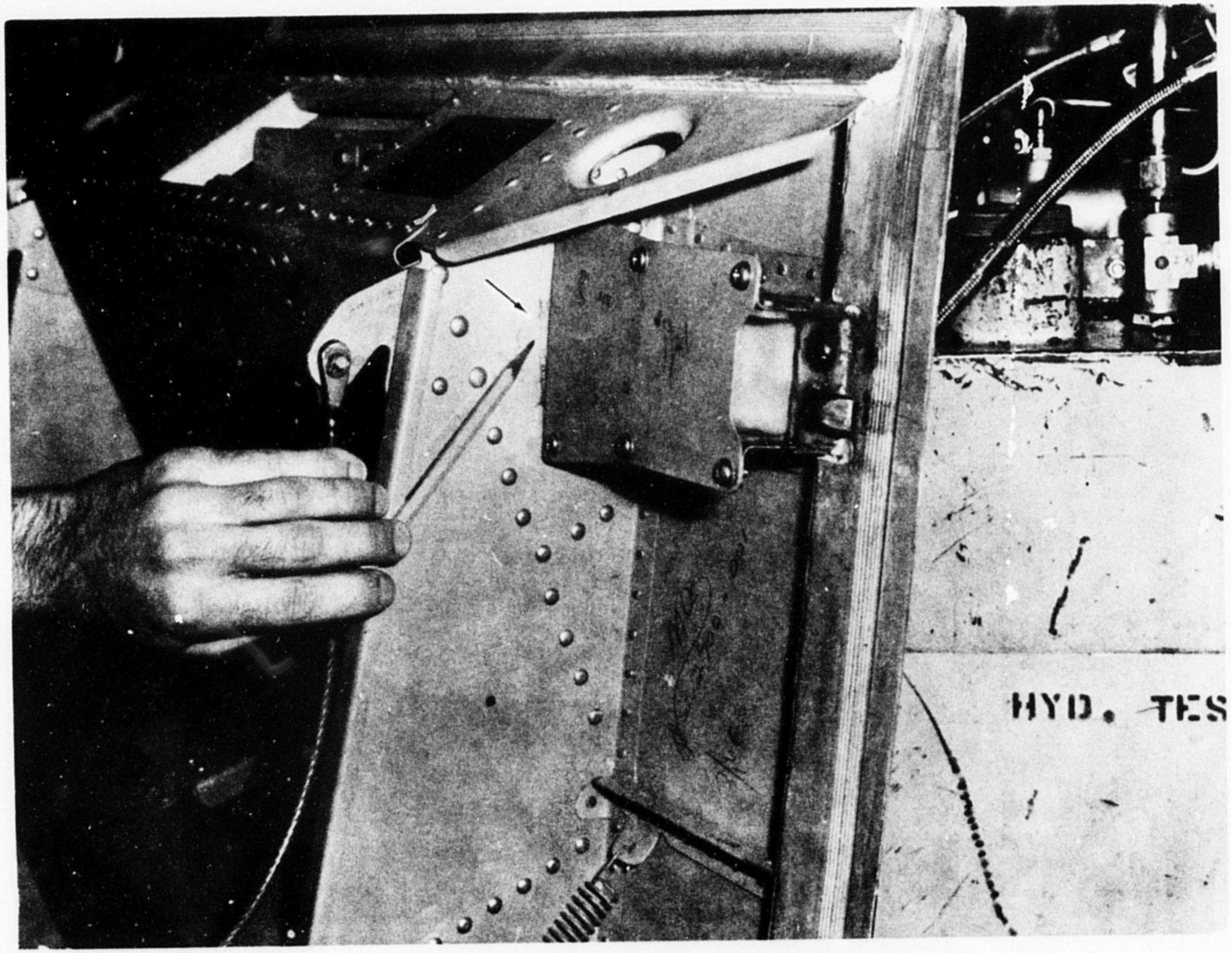


Figure 32. Lower Personnel Door, Side Beam Fatigue Crack.

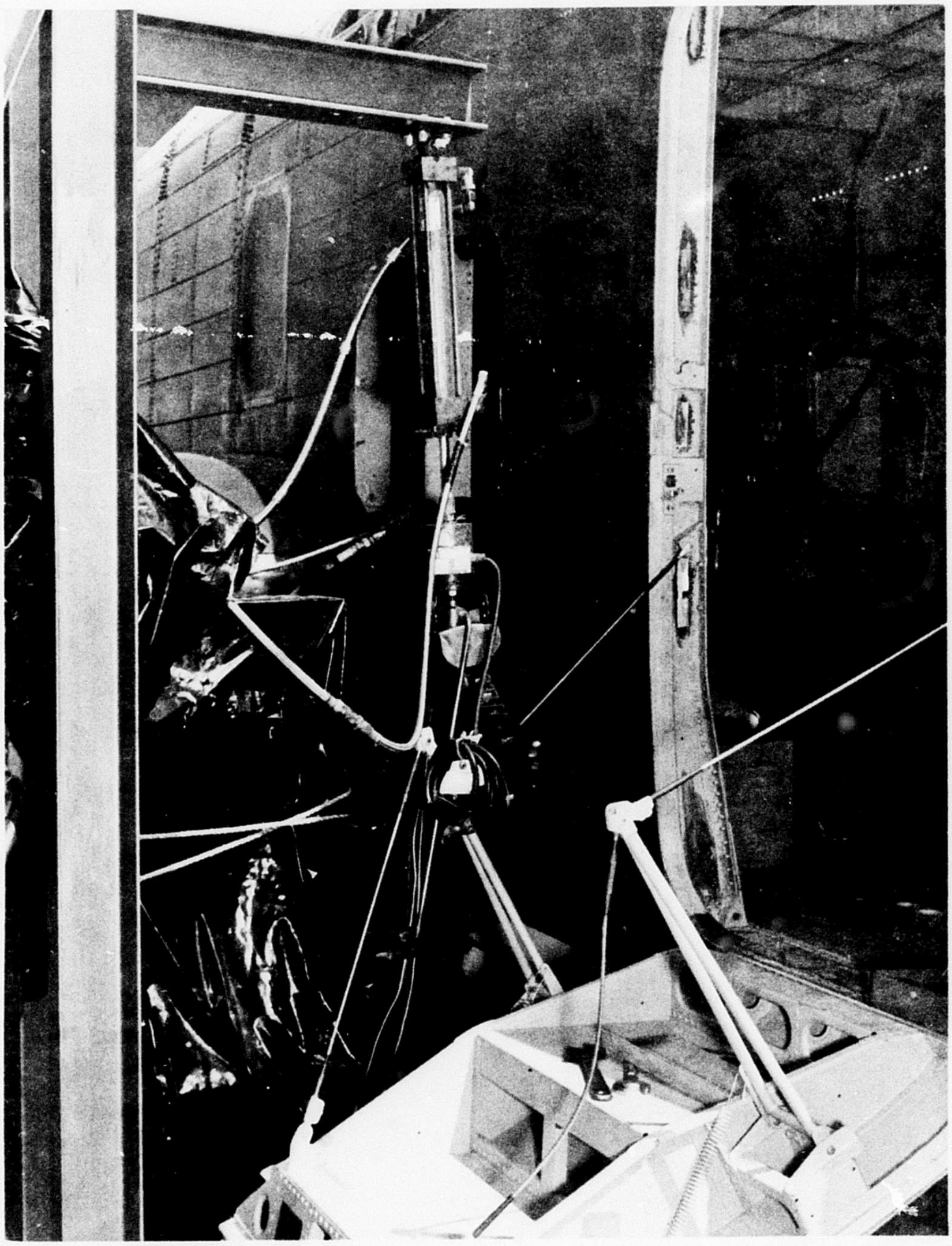


Figure 33. Lower Personnel Door, Cable Impacts.

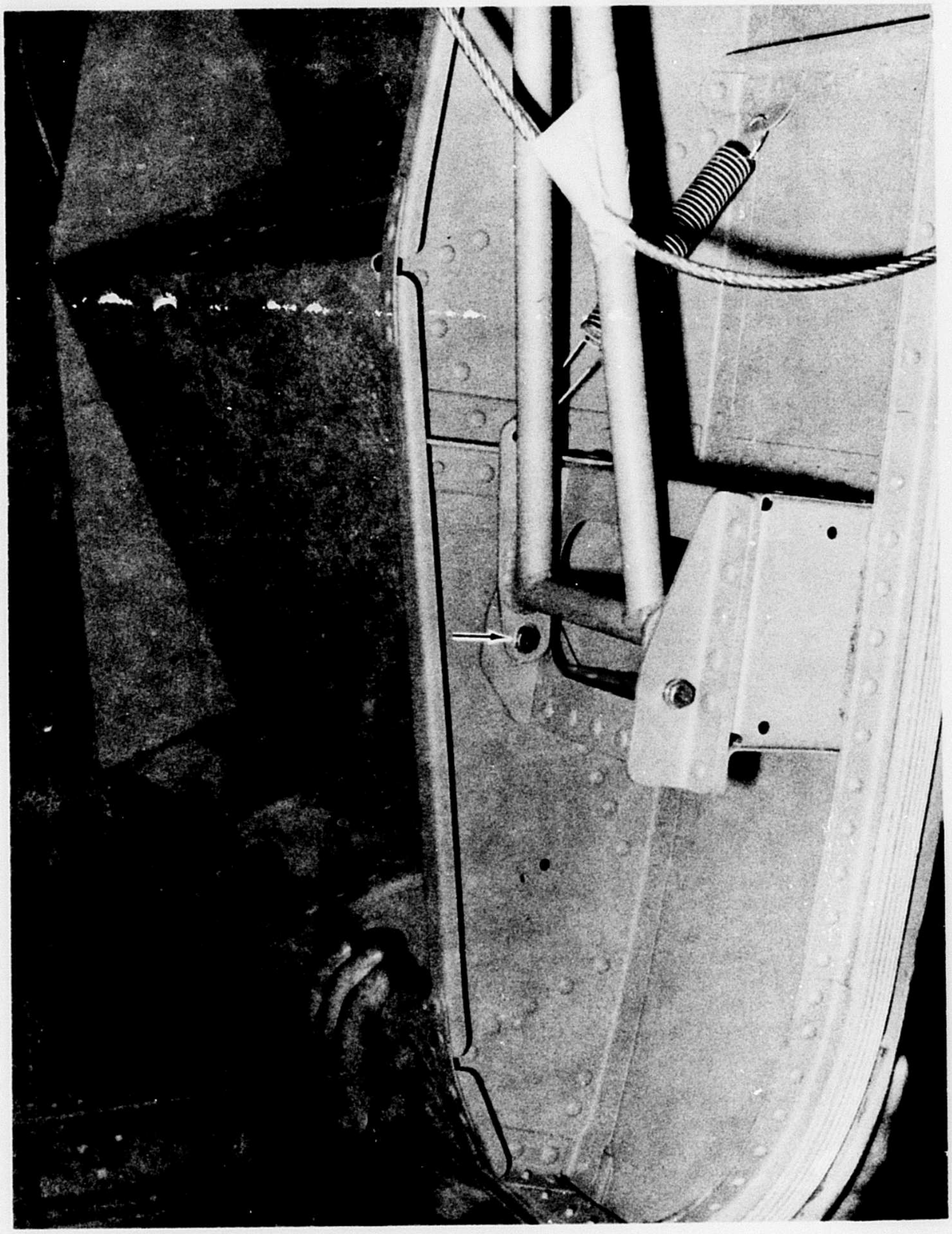


Figure 34. Lower Personnel Door, Original Support Assembly Damage.

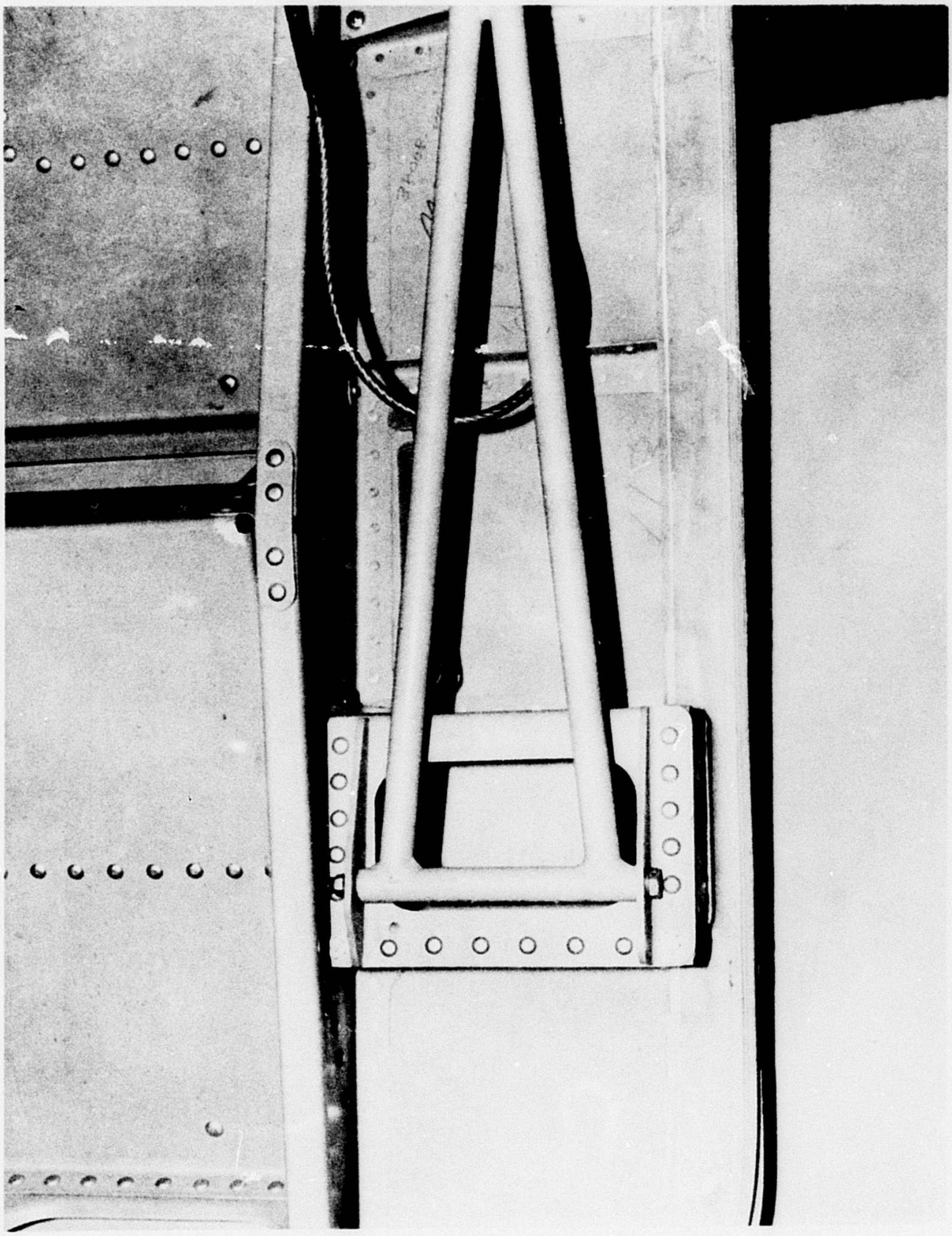


Figure 35. Lower Personnel Door, Redesign, Support Assembly.

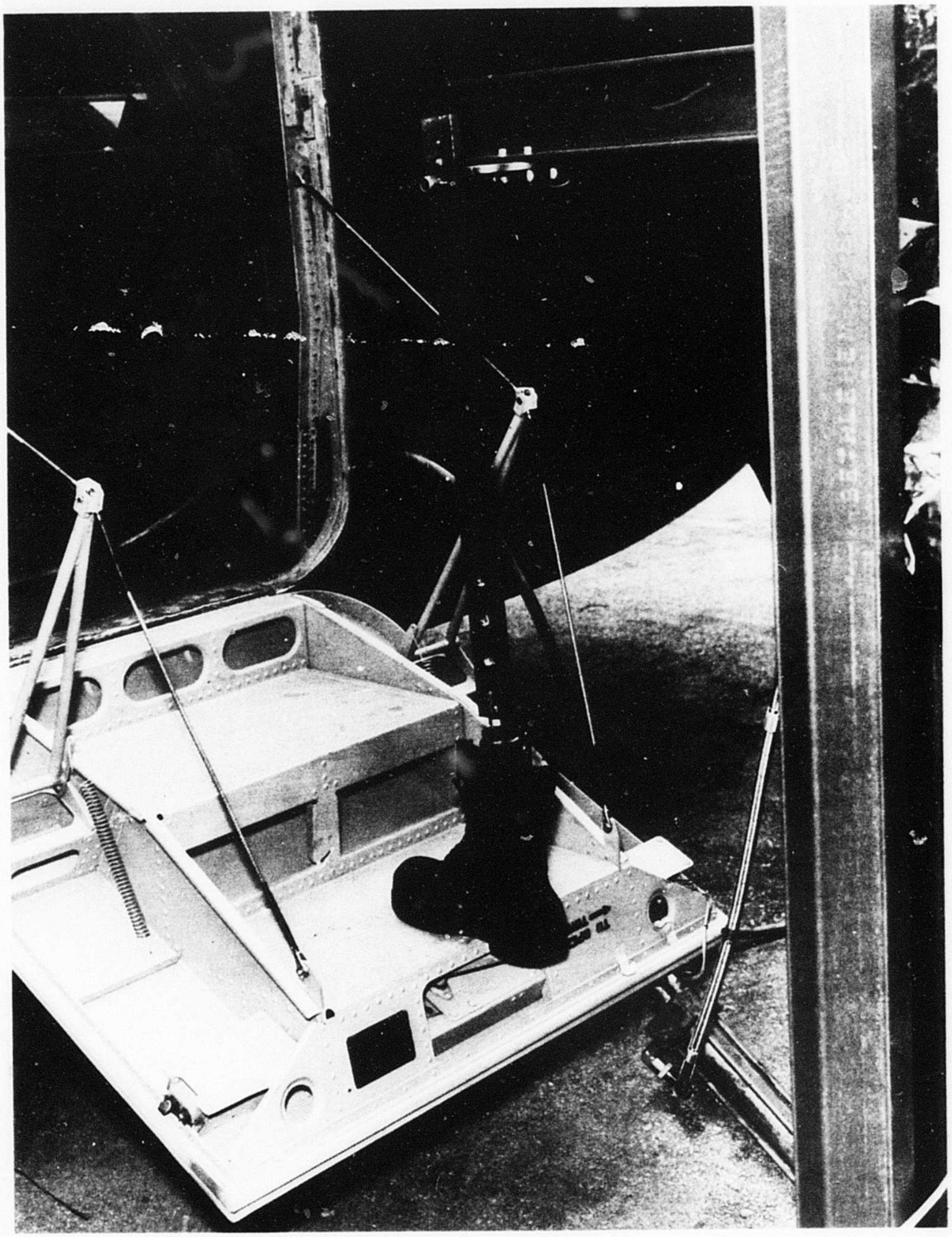


Figure 36. Lower Personnel Door, Riser Kick Test.

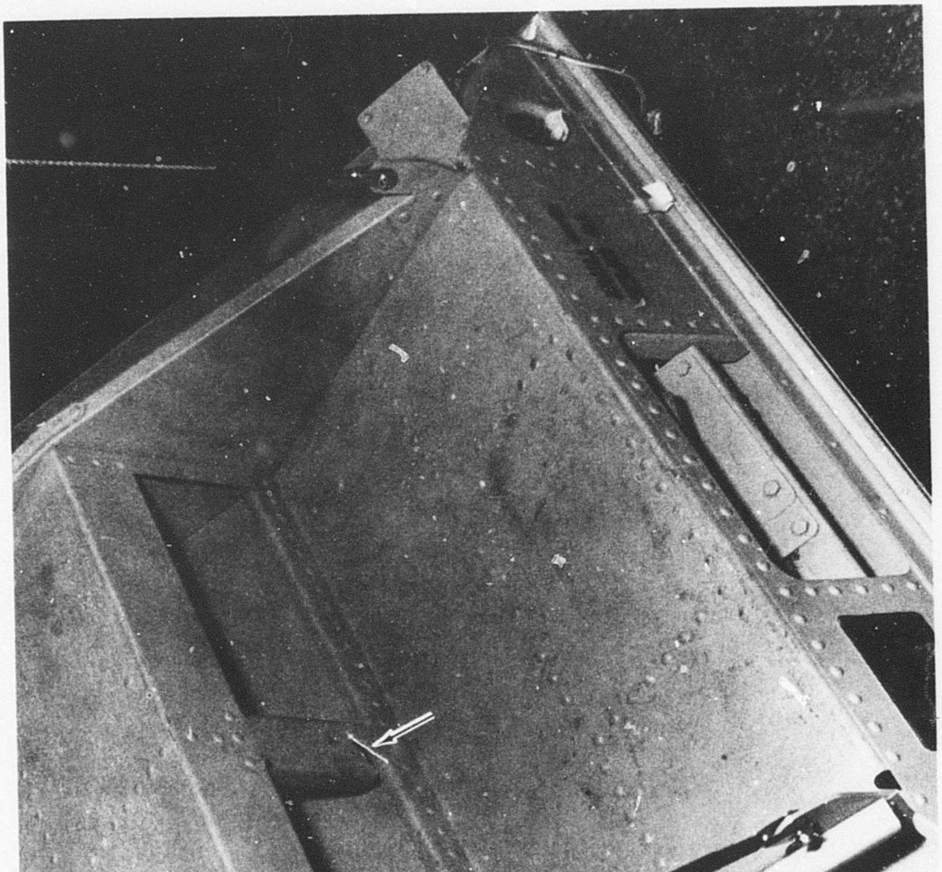


Figure 37. Lower Personnel Door, Original, Riser Fracture.

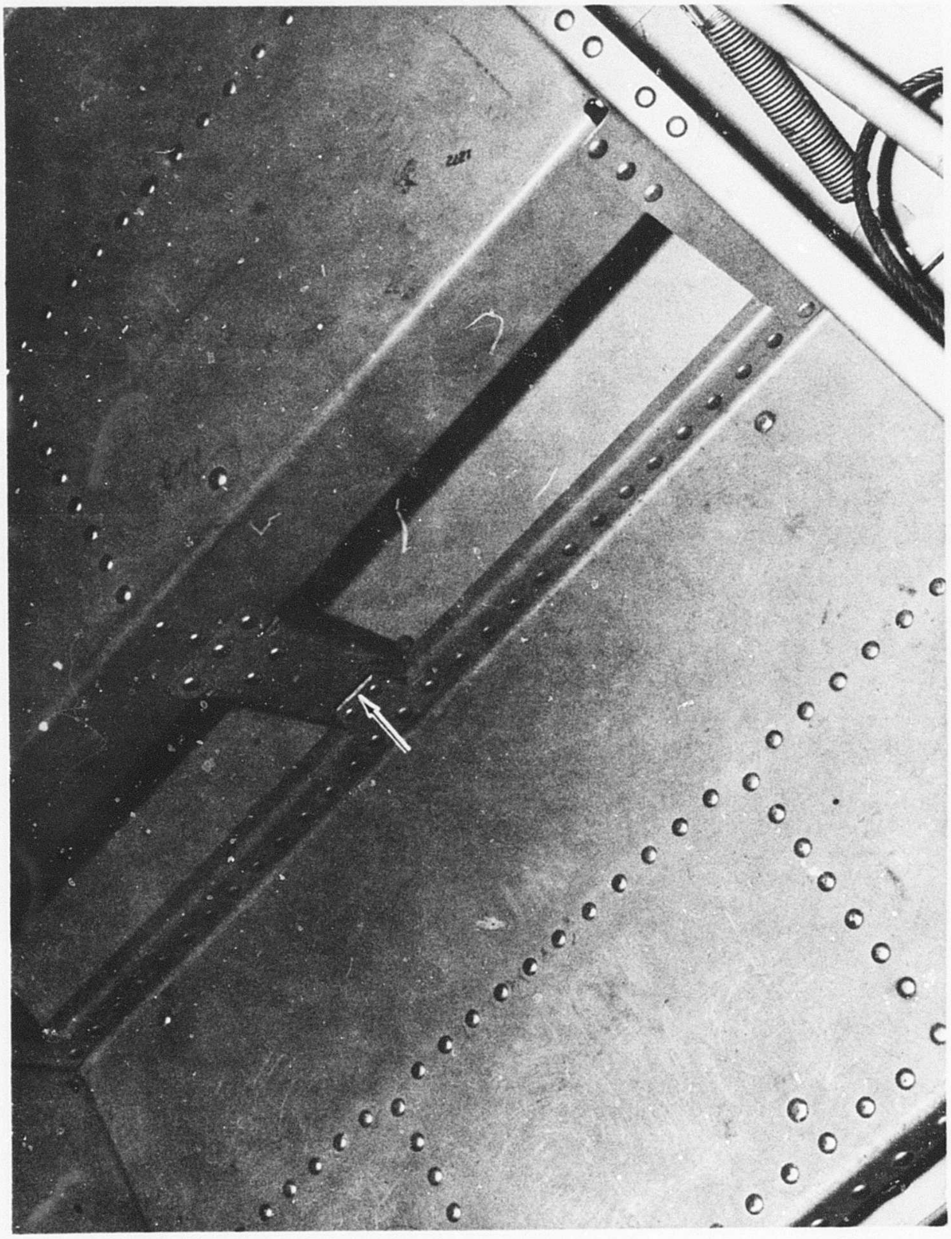


Figure 38. Lower Personnel Door, Redesign, Riser Fracture.

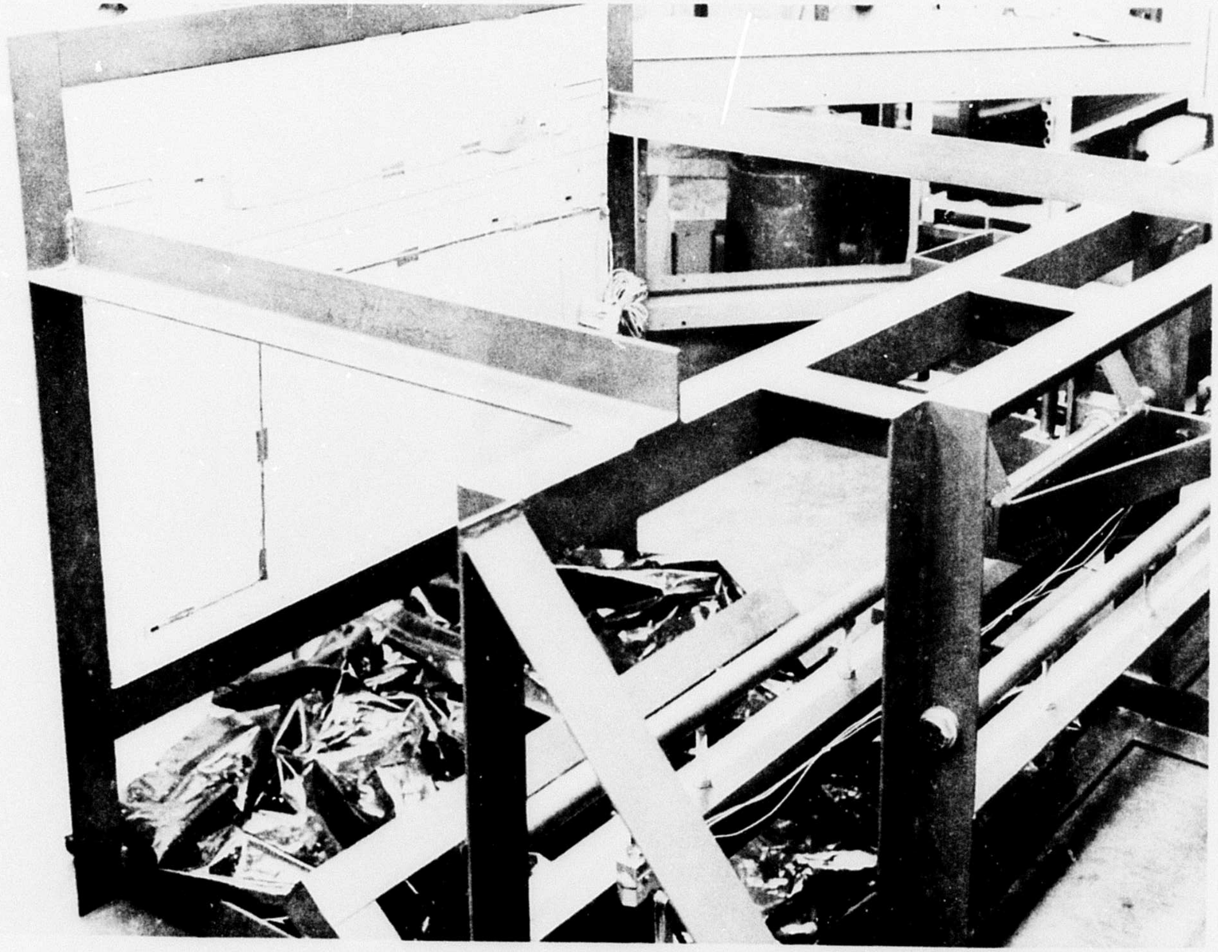


Figure 39. Work Platform Assembly.

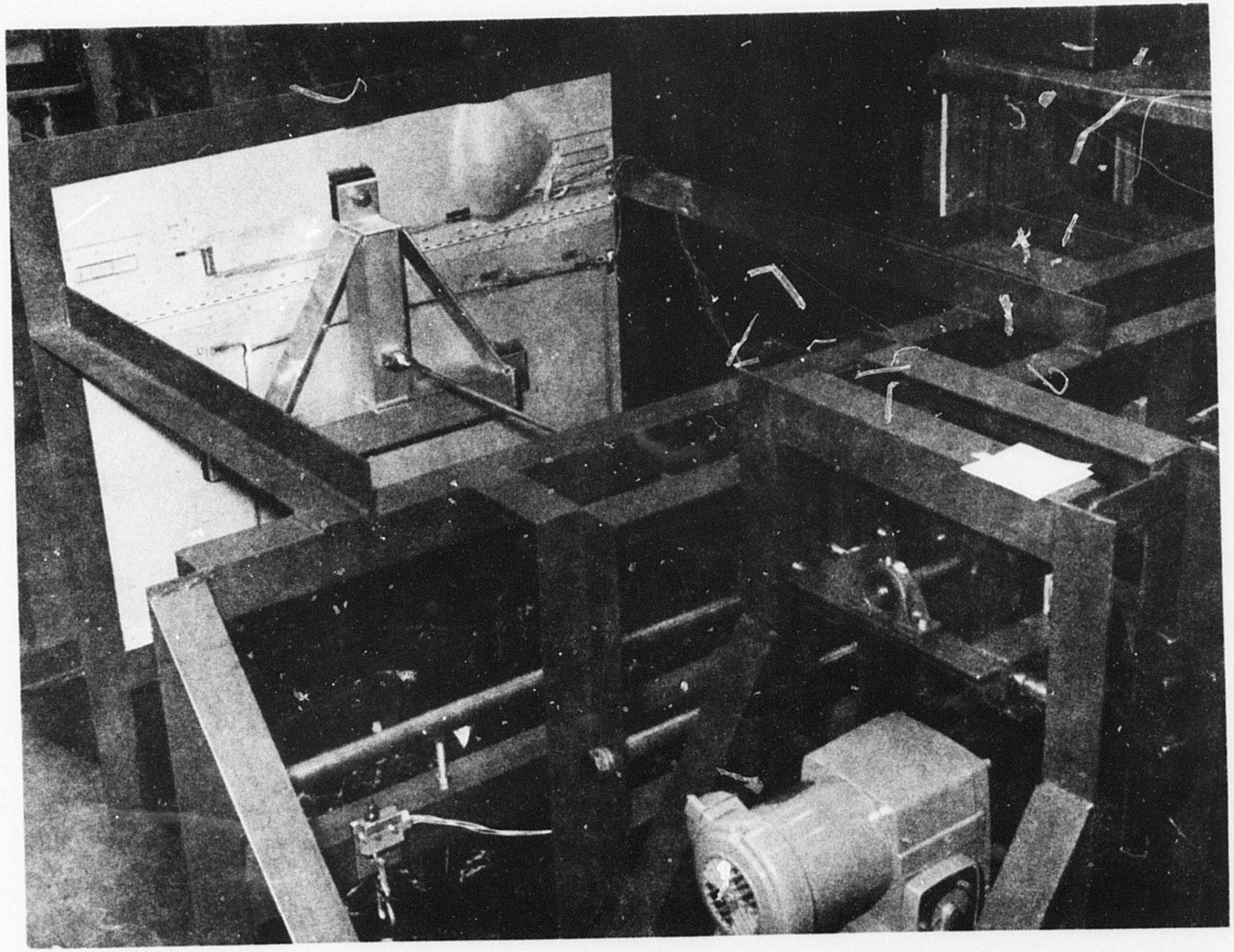


Figure 40. Work Platform Assembly, Vibratory Loading.

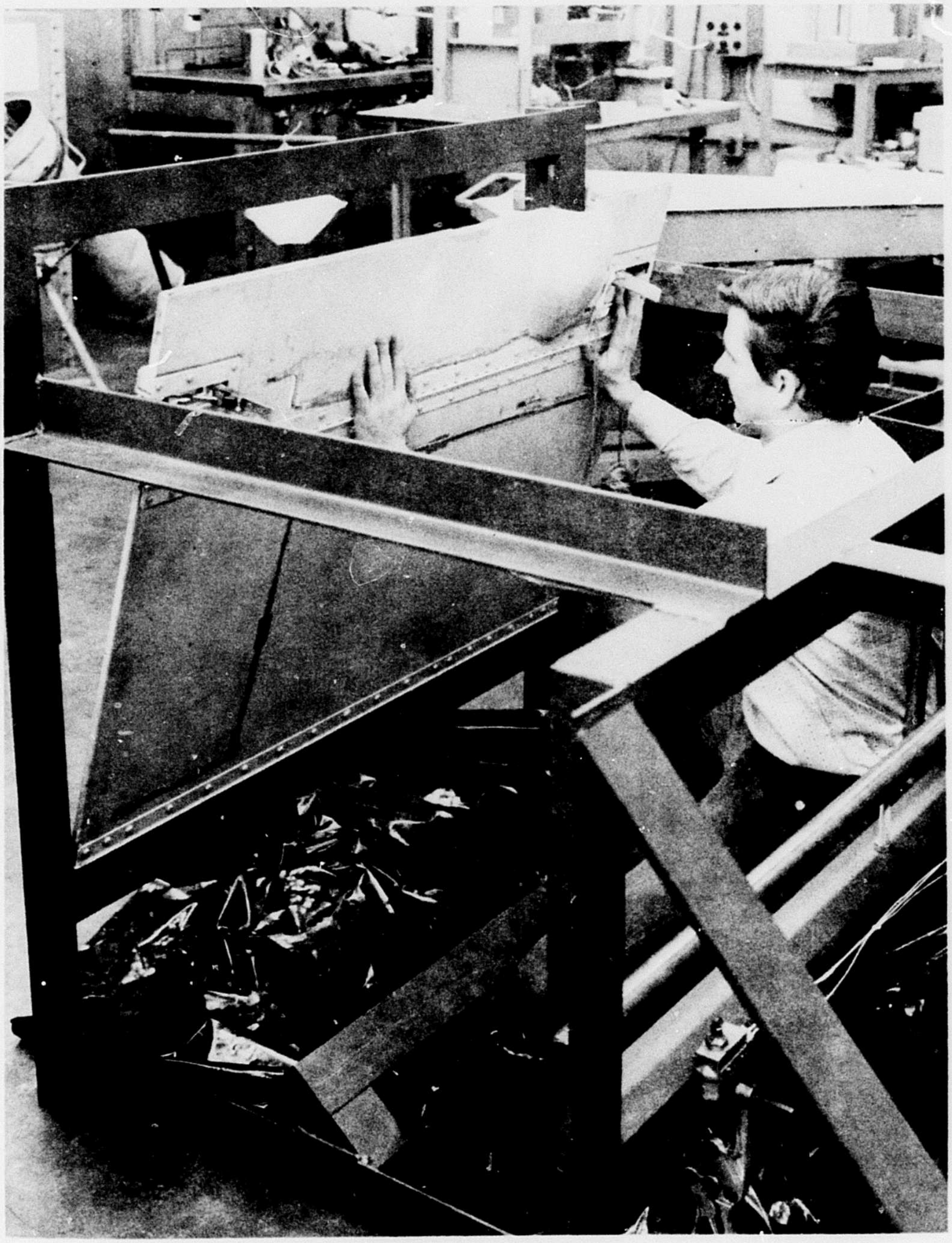


Figure 41. Work Platform Assembly, Manual Cycling.

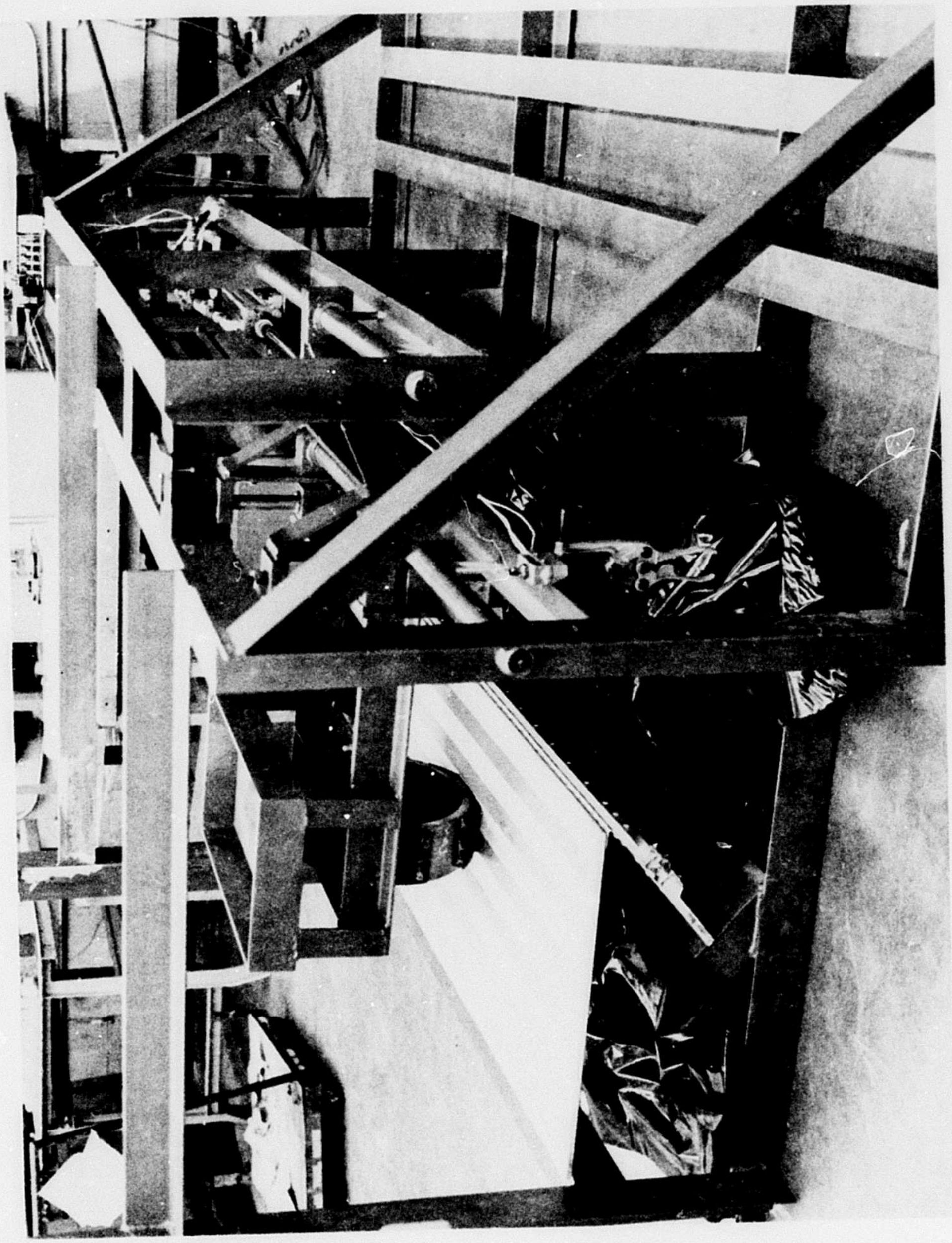


Figure 42. Work Platform Assembly, Roller Load.

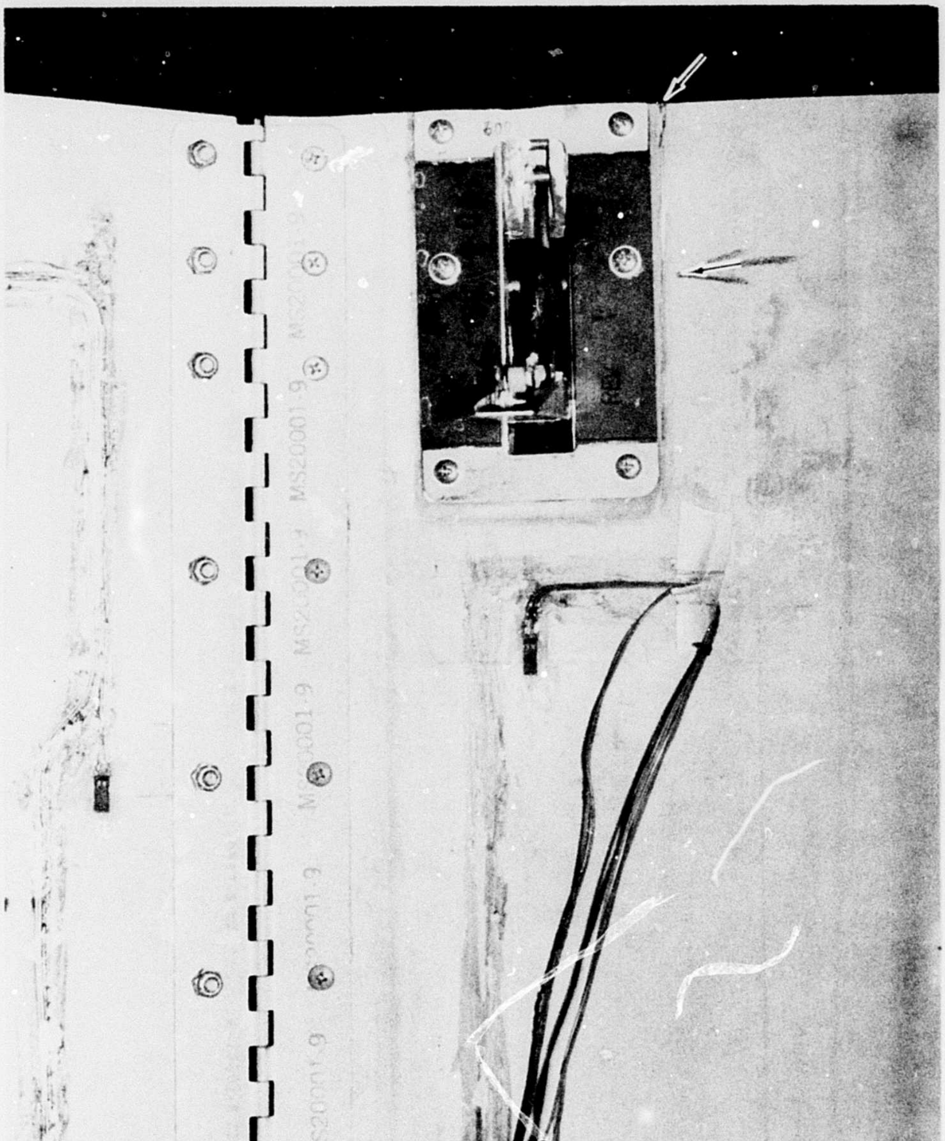


Figure 43. Work Platform Assembly, Original, Crack.

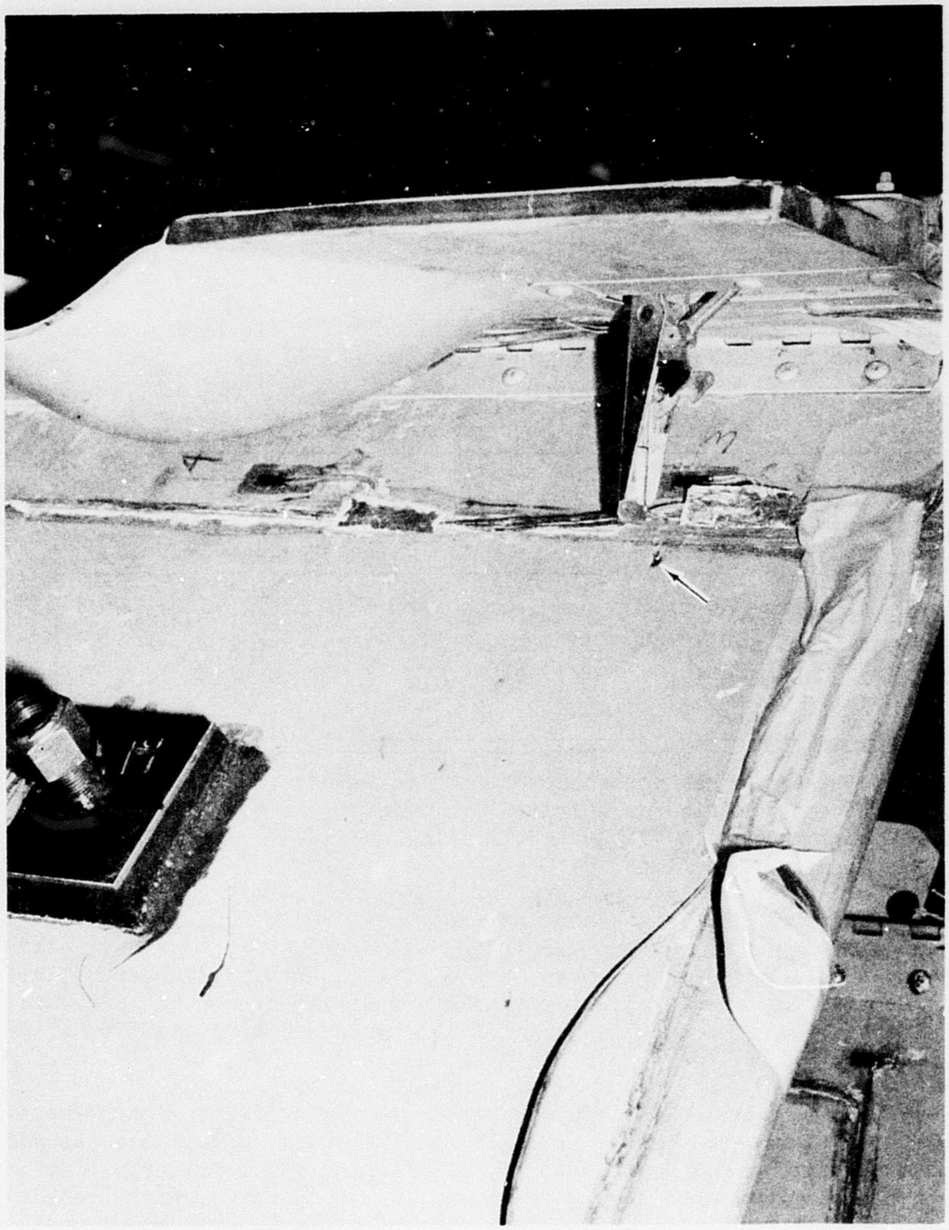


Figure 44. Work Platform Assembly, Redesign, Damage.

One damage mode which appeared on the redesigned work platform assembly, but not on the original, was the cutting through of the structure by the latch handles in the open, working position (Figure 44).

DESIGN TRADE-OFFS

In this study, the three secondary structures that have been the most troublesome in H-53 aircraft from a maintainability standpoint were selected. Design changes were made to improve the mantainability of these structures, and a limited amount of testing was carried out (one specimen each on the original and the redesign) to establish the degree of improvement accomplished. A preliminary trade-off is made of the life-cycle cost effectiveness of these maintainability improvements in relation to the changes in nonrecurring and recurring cost to produce and the change in weight. No attempt has been made to evaluate the added cost involved in designing and testing to more stringent criteria nor the cost reduction resulting from fewer spares for a superior design.

The cost/attribute sensitivites of these factors for the H-53 medium-assault helicopter mission are as shown in Table XVII, assuming a 10-year life cycle.

TABLE XVII. COST SENSITIVITIES

Attribute	Unit	Sensitivity ($/Unit)
Weight Empty	Pound	89.7
Maintainability	MMH/FH	106,800
Nonrecurring Cost	$1000	10
Recurring Cost	$1000	1,660

The delta changes in each of these attributes have been estimated and multiplied by the appropriate sensitivity to arrive at the net life-cycle cost effect on each structure (Tables XVIII, XIX, and XX).

For the hinge and cover and the personnel door, the arbitrary assumption has been made that the maintenance man-hours are reduced in proportion to the number of modes of failure eliminated (as demonstrated by test) divided by the number of modes experienced in the field. This preliminary, limited, cost effectiveness study indicates that the redesign of the hinge cover is cost effective and that the redesign of the personnel door might be slightly cost ineffective.

The test program was unsuccessful in demonstrating a product improvement for the work platform. In the analysis, the arbitrary assumption was made that the redesign was 80% effective in eliminating field failures. It can be seen that improvements have been made at a considerable weight penalty, which would seem to put the redesigned work platform in a doubtful cost-effectiveness category.

TABLE XVIII. HINGE AND COVER ASSEMBLY LIFE-CYCLE COST CHANGE			
	A Unit Change	B Sensitivity	C = A x B Life-Cycle Cost Change
Weight	2.93	89.7	+$ 263
Maintainability	.0099	106,800	- 1,058
Nonrecurring Cost	5.060	10	+ 51
Recurring Cost	.352	1,660	+ 584
Total (Net Change)			-$ 160

TABLE XIX. PERSONNEL DOOR ASSEMBLY LIFE-CYCLE COST CHANGE			
	A Unit Change	B Sensitivity	C = A x B Life-Cycle Cost Change
Weight	.835	89.7	+$ 75
Maintainability	.0045	106,800	- 480
Nonrecurring Cost	10.8	10	+ 108
Recurring Cost	.204	1,660	+ 338
Total (Net Change)			+$ 41

TABLE XX. WORK PLATFORM LIFE-CYCLE COST CHANGE			
	A Unit Change	B Sensitivity	C = A x B Life-Cycle Cost Change
Weight	2(6.81)	89.7	+$ 1,223
Maintainability	.0045	106,800	- 993
Nonrecurring Cost	4.6	10	+ 460
Recurring Cost	.10	1,660	+ 166
Total (Net Change)			+$ 856

CONCLUSIONS

Conclusions of the test are:

(1) Reliability and maintainability techniques (such as FMERA, use of the data bank and trade-offs) are helpful in minimizing problems with secondary structure.

(2) Current field maintenance data are not providing adequate detailed information on damage. Good data could reveal inherent design defects and the exact nature (not vague descriptions) of damage and failures.

(3) Current design and test criteria for secondary structures are not adequate in some areas.

(4) Design and testing of secondary structure must be carefully thought out, as field abuse does not occur in an easily predictable manner.

(5) Major design problems can be minimized by simple functional tests.

RECOMMENDATIONS

As a result of this study, it is recommended that:

1. Design and test specifications for helicopter secondary structures be revised to include functional use and abuse loading conditions associated with maintenance, in addition to operational/flight loads.

2. A study be made of the practicability of requiring that field data collection systems require more descriptive/illustrative/photographic detail as to modes and locations of failures.

3. Reliability and maintainability analytical techniques, such as data bank use, the failure mode and effects analysis and trade offs be applied to helicopter secondary structure as well as to primary structure and mechanism design.

4. Reliability and maintainability be traded off in relation to such other factors as weight, nonrecurring costs, and recurring costs.

APPENDIX I

PRELIMINARY SPECIFICATION REVISIONS RECOMMENDATIONS

PROBLEMS (HINGE AND COVER ASSEMBLY - MAIN ROTOR PYLON)

1. Cover assembly fiber glass cracking and delaminating, distorting and breaking.

2. Latch installation distorting and breaking.

REQUIREMENTS PERTINENT TO ABOVE PROBLEMS

Requirements	Adequate	Need Rev	Nonexistent	Comments
1. General Design				
Army				
AMCP 706-202			X	Doc not issued
Navy				
SD-24H, Vol II				Superseded by SD-24J, Vol II
Fiber Glass			X	See Note 1
3.2.4.2.4		X		See Note 2
SD-24J, Vol II				
3.2.4.1.5	X			See Note 6
3.2.4.2.4		X		See Note 2
Air Force				
AFSC DH 2-1				
DN 2A1		X		See Note 3
DN 3A3			X	See Note 4
MIL-I-83294 (USAF)				
3.4.9.5		X		See Note 5
4.2	X			
Sikorsky				
SS 9583		X		See Note 6
2. Structural Design				
Army, Navy, Air Force				
MIL-S-8698 (ASG) (-1)				Superseded by AR-56

Requirements	Adequate	Need Rev	Nonexistent	Comments
3.1.3.3 and 3.1.3.4		X		See Note 7
3.1.9	X			
3.2.2.2		X		See Note 8
AR-56				Supersedes MIL-S-8698
3.1 - 3.1.4	X			
3.1.9 - 3.2.1.3		X		See Note 9
3.4.7	X			
3. Test, Ground				
Army, Navy, Air Force MIL-T-8679				
3.1.10.7		X		See Note 10
3.2.9.1.2	X			
3.2.9.3.4	X			
Army AMCP 706-203				
2 - 2.2	X			
9 - 2.2 thru 9 - 2.2.1, Test condition 15	X			
Air Force AFSC DH2 - 1				
DN 2A1	X			
MIL-I-83294 (USAF)				
4.7	X			
4.7.3	X			
4. Demonstration				
Army AMCP 706-203				
2 - 2.2	X			
9 - 14	X			
10 - 1	X			
10 - 2.1	X			
Navy MIL-D-23222A(AS)				

Requirements	Adequate	Need Rev	Nonexistent	Comments
3.17 - 3.17.1	X			
Air Force AFSC DH 2-1 DN 2A1	X			

NOTES

1. SH-24H, Vol II, does not contain requirements for fiber glass; however, it is superseded by SH-24J, Vol II, which does (in para. 3.2.4.1.5).

2. SD-24H, Vol II, para. 3.2.4.2.4 has been superseded by SD-24J, Vol II, para. 3.2.4.4. The latter should be revised through the addition of "movable sections" throughout (e.g., doors, movable sections, and removable sections). In addition, "transmission, rotor head" should be inserted following "engine" in the first sentence (e.g., engine, transmission, rotor head, and accessories).

3. AFSC DH 2-1, DN 2A1, para. 4, Dynamic Loads - Add as applicable: h. Operations of doors, work platforms, movable or removable covering or fairing, cowlings, etc., during loading, boarding, inspection, and maintenance operations.

4. AFSC DH 2-1, DN 3A3, para. 3, Fairing - Add: If fairing is of the sliding type (e.g., used on helicopters for access to transmissions or rotor head), it should meet the deformation and fatigue requirements of MIL-T-8679, para. 3.1.10.7.

5. MIL-I-83294 (USAF), para. 3.4.9.5. If this specification is intended for application to helicopters, this paragraph should be expanded, or a new one added to include fairing (cowling) requirements for transmissions and rotor heads.

6. As required by SD-24J, Sikorsky submitted SS 9583. This Sikorsky specification, previously approved by the Government, is being revised to require materials with improved interlaminar shear strength not currently provided in military specification materials. Sikorsky will request that the military specification also be revised in the near future. This should reduce delamination problems to a minimum. However, in order to eliminate delamination and cracking of fiber glass, good judgment must be exercised in designs that are subject to penetration (dropped tools, etc.) and localized pounding (due to vibration or repeated loads resulting from normal looseness of slide or roller type installations, latches, etc.) to determine if fiber glass is suitable for the application, if metal reinforcement is required, or if metal should be used instead. Fiber glass might delaminate or crack under such conditions; metal would probably yield instead. (We recommend that the Army include such design information in AMCP 706-202 when issued.)

7. MIL-S-8698 (ASG) (-1), para. 3.1.3.3. Expand to include "movable or removable covering or fairing."

8. MIL-S-8698 (ASG) (-1), para. 3.2.2.2. Add: Design fatigue loading for movable or removable covering or fairing shall include loads and effects of abuse (slamming, forcing, etc.) imposed by personnel during inspection and maintenance of the aircraft.

9. AR-56, para. 3.1.9.1. Add requirement similar to that added in Note 8 above.

10. MIL-T-8679, para. 3.1.10.7. Change to read: Deformation and fatigue of doors, work platforms, movable or removable covering of fairing, cowling, locks, latches, slides, rollers, and fasteners - It shall be shown during structural tests that these items and items of mechanical equipment, such as landing gear, remain in their intended positions consistent with specified structural design requirements. It shall also be shown that the following fatigue or repeated load tests have been met:

Item	Open/Close Cycles	Repeated Force	Impact Cycles
Door, Entrance			
a. with stairs			
b. without stairs			
Door, Inspection			
a. hinged			
Platforms, Work			
a. operable			
b. fixed			
Cowling/Covering/Fairing			
a. removable			
b. hinged			
c. sliding			

NOTE: Cycles and loads are peculiar to aircraft model.

DOOR, EXTERIOR, PERSONNEL
(Part Number 65207-03006-011)

PROBLEMS

1. Lower door, stairs (steps) cracking, exterior skin cracking, distortion, dents or bending.

2. Support assembly, weak at attaching points, cracking.

3. Cable assembly, strands breaking.

4. Latch installation, distortion, wear, breaking.

5. Hinge, cracking.

REQUIREMENTS PERTINENT TO ABOVE PROBLEMS

Requirements	Adequate	Need Rev	Nonexistent	Comments
1. General Design				
Army				
AMCP 706-202			X	Doc not issued
Navy				
SD-24H, Vol II				Superseded by SD-24J, Vol II
3.7.1.6	X			
3.7.1.6.1	X			
3.7.1.7.1	X			
SD-24J, Vol II				
3.7.1.6	X			
3.7.1.6.1	X			
3.7.1.7.1	X			
Air Force				
AFSC DH 2-1				
DN 2A1		X		See Note 1
AFSC DH 2-2				
DN 2A1 (5.7)	X			
2. Structural Design				
Army, Navy, Air Force				
MIL-S-8698 (ASG) (-1)				

Requirements	Adequate	Need Rev	Nonexistent	Comments
3.1.3.3 and 3.1.3.4	X			
3.1.9.	X			
3.2.2.2		X		See Note 2
AR-56				
3.1 - 3.1.4	X			
3.1.9 - 3.2.1.3		X		See Note 3
3.4.7 - 3.4.8	X			
3. Test, Ground				
Army, Navy, Air Force MIL-T-8679				
3.1.10.7		X		See Note 4
3.2.9.3.4	X			
Army AMCP 706-203				
2 - 2.2	X			
9 - 2.2 thru 9 - 2.2.1, test condition 15	X			
9 - 10.2.6	X			
Air Force AFSC DH 2-1				
DN 2A1	X			
4. Demonstration				
Army AMCP 706-203				
2.2.2	X			
9 -14	X			
10 - 1	X			
10 - 2.1	X			
Navy MIL-D-23222A (AS)				
3.17 - 3.17.1	X			
Air Force AFSC DH 2-1				
DN 2A1	X			

NOTES

1. AFSC DH 2-1, DN 2A1, para. 4, Dynamic Loads - Add as applicable: h. Operation of doors, work platforms, movable or removable covering or fairing, cowlings, etc., during loading, boarding, inspection, and maintenance operations.

2. MIL-S-8698 (ASG)(-1), para. 3.2.2.2. - Add: Design fatigue loading for doors, boarding steps, and combinations thereof shall include loads and effects of abuse (slamming, jumping, kicking, etc.) imposed by personnel during loading, boarding, and inspection and maintenance of the aircraft.

3. AR-56, para. 3.1.9.1. Add requirement similar to that added in Note above.

4. MIL-T-8679, para. 3.1.10.7. Change to read: Deformation and fatigue of doors, work platforms, movable or removable covering or fairing, cowling, locks, latches, slides, rollers, and fasteners. It shall be shown during structural tests that these items and items of mechanical equipment, such as landing gear, remain in their intended positions consistent with specified structural design requirements. It shall also be shown that the following fatigue or repeated load tests have been met:

Item	Open/Close Cycles	Repeated Force	Impact Cycles
Door, Entrance			
a. with stairs			
b. without stairs			
Door, Inspection			
a. hinged			
Platforms, Work			
a. operable			
b. fixed			
Cowling/Covering/Fairing			
a. removable			
b. hinged			
c. sliding			

NOTE: Cycles and loads are peculiar to aircraft model.

PLATFORMS, MAINTENANCE (WORK) - MAIN ROTOR PYLON
(Part Number 65207-09004-041, -042)

PROBLEMS

1. Upper and lower fiber glass panels cracking, delaminating, and distorting.
2. Latch assembly breaking.
3. Hinge halves breaking.
4. Hinge pins breaking and working loose.

REQUIREMENTS PERTINENT TO ABOVE PROBLEMS

Requirements	Adequate	Need Rev	Nonexistent	Comments
1. General Design				
Army				
AMCP 706-202			X	Doc not issued
Navy				
SD-24H, Vol II				Superseded by SD-24J, Vol II
3.2.2.2.10	X			
Fiber Glass			X	See Note 1
3.11.7		X		See Note 2
2.23.2.4	X			
SD-24J, Vol II				Supersedes SD-24H, Vol II
3.2.2.2.3.8	X			
3.2.4.1.5	X			See Note 10
3.11.7		X		See Note 2
3.23.2.4	X			
Air Force				
AFSC DH 2-1				
DN 2A1		X		See Note 3
DN 3A3			X	See Note 4
MIL-I-83294 (USAF)				
3.4.9.5		X		See Note 5
4.2	X			
Sikorsky				
SS 9583		X		See Note 10

Requirements	Adequate	Need Rev	Nonexistent	Comments
2. Structural Design				
Army, Navy, Air Force MIL-S-8698 (ASG) (-1)				Superseded by AR-56
3.1.3.3 and 3.1.3.4		X		See Note 6
3.1.9	X			
3.2.2.2		X		See Note 7
AR-56				Supersedes MIL-S-8698
3.1 - 3.1.4	X			
3.1.9 - 3.2.1.3		X		See Note 8
3.4.7	X			
3. Test, Ground				
Army, Navy, Air Force MIL-T-8679				
3.1.10.7		X		See Note 9
3.2.9.1.2	X			
3.2.9.3.4	X			
Army AMCP 706-203				
2 - 2.2	X			
9 - 2.2 thru 9 - 2.2.1, test condition 15	X			
Air Force AFSC DH 2-1				
DN 2A1	X			
MIL-I-83294 (USAF)				
4.7	X			
4.7.3	X			
4. Demonstration				
Army AMCP 706-203				
2 - 2.2	X			
9 - 14	X			
10 - 1	X			
10 - 2.1	X			

Requirements	Adequate	Need Rev	Nonexistent	Comments
Navy MIL-D-23222A (AS) 3.17 - 3.17.1	X			
Air Force AFSC DH 2-1 DN 2A1	X			

NOTES

1. SD-24H, Vol II, does not contain requirements for fiber glass; however, it is superseded by SD-24J, Vol II, which does (in para. 3.2.4.1.5).

2. SD-24H, Vol II, para. 3.11.7 has been superseded by SD-24J, Vol. II, para. 3.11.7. The latter should be revised to read: 3.11.7 Integral Work Platforms. Integral work platforms shall be provided to permit access to and maintenance of engines, transmissions, and rotor heads that cannot be reached readily from other parts of the aircraft, the ground, or the ship's deck.

3. AFSC DH 2-1, DN2A1, para. 4, Dynamic Loads - Add as applicable: h. Operation of doors, work platforms, movable or removable covering or fairing, cowlings, etc., during loading, boarding, inspection, and maintenance operations.

4. AFSC DH 2-1, DN 3A3 - Add requirement for integral work platforms similar to that in Note 2 above.

5. MIL-I-83294 (USAF), para. 3.4.9.5. If this specification is intended for application to helicopters, this paragraph should be expanded, or a new one added, to include integral work platform requirements for access to transmissions and rotor heads.

6. MIL-S-8698 (ASG)(-1), para. 3.1.3.3. Expand to include integral work platforms.

7. MIL-S-8698 (AG)(-1), para. 3.2.2.2. Add: Design fatigue loading for doors, boarding steps, and integral work platforms shall include loads and effects of abuse (slamming, jumping, kicking, etc.) imposed by personnel during loading, boarding, inspection, and maintenance of the aircraft.

8. AR-56, para. 3.1.9.1. Add requirement similar to that added in Note 7 above.

9. MIL-T-8679, para. 3.1.10.71 - Change to read: Deformation and fatigue of doors, work platforms, movable or removable covering or fairing, cowling, locks, latches, and fasteners. It shall be shown during structural tests that these items and items of mechanical equipment, such as landing gear,

remain in their intended positions consistent with specified structural design requirements. It shall also be shown that the following fatigue or repeated load tests have been met.

Item	Open/Close Cycles	Repeated Force	Impact Cycles
Doors, Entrance			
a. with stairs			
b. without stairs			
Doors, Inspection			
a. hinged			
Platforms, Work			
a. operable			
b. fixed			
Cowling/Covering/Fairing			
a. removable			
b. hinged			
c. sliding			

NOTE: Cycles and loads are peculiar to aircraft model.

10. SS 9583. This Sikorsky specification, previously approved by the Government, is being revised to require materials with improved interlaminar shear strength not currently provided in military specification materials. We will request that the military specification also be revised in the near future. This should reduce delamination problems to a minimum. However, in order to eliminate delamination and cracking of fiber glass, good judgment must be exercised in designs that are subject to penetration (dropped tools, hard heels, etc.) and localized pounding (due to vibration or repeated loads resulting from normal looseness of latches and other fasteners) to determine if fiber glass is suitable for the application, if metal reinforcement is required, or if metal should be used instead. Fiber glass might delaminate or crack under such conditions; metal would probably yield instead. (We recommend that the Army include such design information in AMCP 706-202 when issued.)

APPENDIX II

FAILURE MODE EFFECT AND RELIABILITY ANALYSIS

This appendix provides:

(1) Reliability Logic Diagrams

(2) Failure Mode and Effect Analyses

(3) Reliability Analysis

for the following secondary structures:

a. Main Rotor Pylon Fairing Housing Assembly

b. Main Rotor Pylon Fairing Hinge and Cover Assembly

c. Main Rotor Pylon Fairing Slide and Cover Assembly

d. Cockpit and Canopy Door Installation, Nose Gear

e. Fuselage Door Installation

f. Sponson Cover Installation, Fuel Cell

g. Sponson Platform Assembly, Service Platform

h. Main Rotor Pylon Fairing Platform Assembly, Work Platform

i. EAPS, Rear Frame Assembly

j. Tail Boom Support Installation, Compass Transmitter

TABLE XXI. RELIABILITY LOGIC DIAGRAM -
MAIN ROTOR PYLON FAIRING HOUSING ASSEMBLY

Diagram No. 1

HOUSING ASSY #10
65205-09006-044

HOUSING #11
65205-09006-105

HAND GRIP (2 REQUIRED) #12
65209-09011-101

PANEL, OIL COOLER ACCESS #13
65207-09008-011

ATTACHING SCREWS AN525-10R-8
STIFFENERS EVERY 8-10 INCHES IN 3 DIMENSIONS, EXCEPT PANEL
3 EXTRA PLIES AT CORNERS
2 EXTRA PLIES AT STIFFENERS

Probable Areas of Failure for Housing Assy

1. Delamination at Any Edge
2. Cracking at Any Screw or Rivet Hole, or Close to Hand Grip
3. Puncture From Maintenance Personnel Walking on Housing

TABLE XXII. MAIN ROTOR PYLON FAIRING HOUSING ASSEMBLY FAILURE MODE AND EFFECTS ANALYSIS

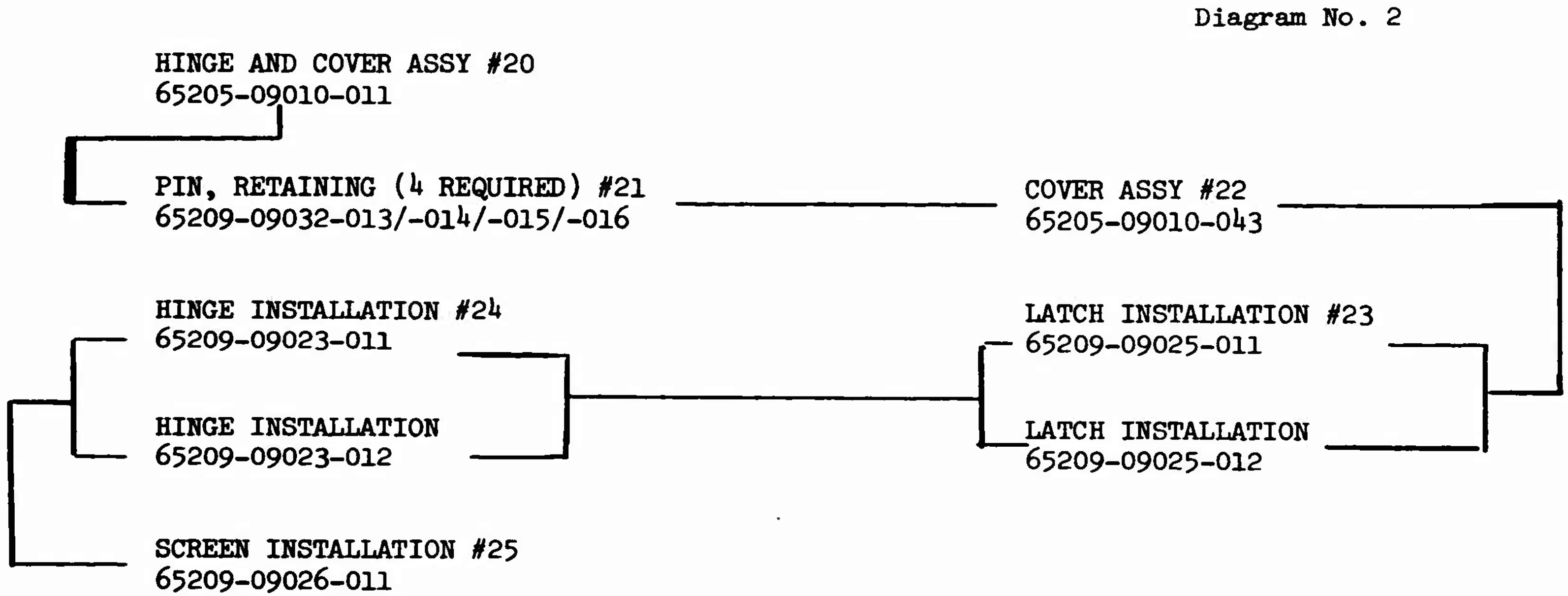
ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Housing	11	65205-09006 -105	1	Aerodynamic Fairing	Delamination	INFLIGHT	Loss of structural integrity	Loss of structural integrity of housing assy.	Damage to blades & fuselage, significantly increased drag	Visual inspection on landing or audio report or unusual vibration in flight	Immediate corrective action is not critical to A/C safety	3 extra plies at corners, 2 extra plies at stiffeners. Stiffeners every 8 - 10'	Attached by screws to adjacent housing & fuselage
					Cracking	SAME AS DELAMINATION							
				Work Platform (Not intended function)	Puncture	Inspection or Maint.	SAME AS DELAMINATION						
Hand Grip	12	65209-09011 -101	1	Handhold	Delamination	Inspection or Maint.	Minor safety hazard to maintenance personnel from sharp edges	Loss of structural integrity of housing assy.	None	Visual Inspection	Same as above		Riveted to housing, likely area for housing failure
					Cracking	SAME AS DELAMINATION							
Panel, Oil Cooler Access	13	65207-09008 -011	1	Air Inlet to Oil Cooler	Cracking	INFLIGHT	Loss of structural integrity	None	Damage to blades & fuselage. Increased drag.	Visual Inspection	Same as above	2 extra plies in attachment area	Attached by screws to housing
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8a)	(8b)	(8c)	(9)	(10)	(11)	(12)

NOTE: FAILURES THAT ARE THE RESULT OF QUALITY CONTROL, FOR EXAMPLE, BLISTERING OR TACKINESS, OR FAILURES THAT DO NOT REQUIRE REMOVAL OR REPAIR OF THE PART, SUCH AS ABRASION OR CRAZING, WERE NOT CONSIDERED.

TABLE XXIII. MAIN ROTOR PYLON FAIRING HOUSING ASSEMBLY RELIABILITY ANALYSIS

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO/ FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE		FAILURE MODE RATIO α	ENVIRONMENTAL RATIO K_E	OPERATIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/ HOUR λ_G		FAILURE MODE CONTRIBUTION ($\alpha K_E K_A \cdot \lambda_G$)	COMPONENT CRITICALITY NO., C_r
Housing Assy.	10	65205-09006 -044	1	Aerodynamic Fairing	Cracking Delamination Puncture Misc.	All	Component actual loss of function requiring repair or replacement	3-M Report 24 July '72 H/SH-3A/D/G WUC 1122F, 11230 How malf. SH-3A 9/67-10/68		.571 .050 .020 .359	1.000 1.000 1.000 1.000	1.000 2.000 2.000 1.000	.0036 .0036 .0036 .0036		.0021 .0004 .0002 .0013	.0040
Housing	11	65205-09006 -105		Aerodynamic Fairing	Delamination	INFLIGHT			Hazard Level II	Possible	CRACKING MOST LIKELY AROUND HAND GRIPS AND ATTACHING SCREW HOLES					
					Cracking	INFLIGHT			II	Probable						
				Work Platform	Puncture	Inspection & Maint.			I	Possible						
Hand Grip	12	65209-09011 -101		Handhold	Delamination	Inspection & Maint.			I	Not very possible						
					Cracking	Inspection & Maint.			I	Possible						
Panel, Access	13	65207-09008 -011		Air Inlet to Oil Cooler	Cracking	INFLIGHT			II	Probable	CRACKING OF SCREEN WEBS LIKELY					
				(13)	(14)	(15)	(16)	(17)	(26) (18)	(19)	(20)	(21)	(22)	(23)	(24)	(25)

TABLE XXIV. RELIABILITY LOGIC DIAGRAM - HINGE AND COVER ASSEMBLY



ALUMINUM STIFFENERS EVERY 7½ INCHES

Probable Areas of Failure

1. Cracking of Aluminum Stiffeners
2. Breaking or Distortion of Latches
3. Delamination of Skin at Edges
4. Cracking of Skin

TABLE XXV. MAIN ROTOR PYLON FAIRING HINGE AND COVER ASSEMBLY FAILURE MODE AND EFFECTS ANALYSIS

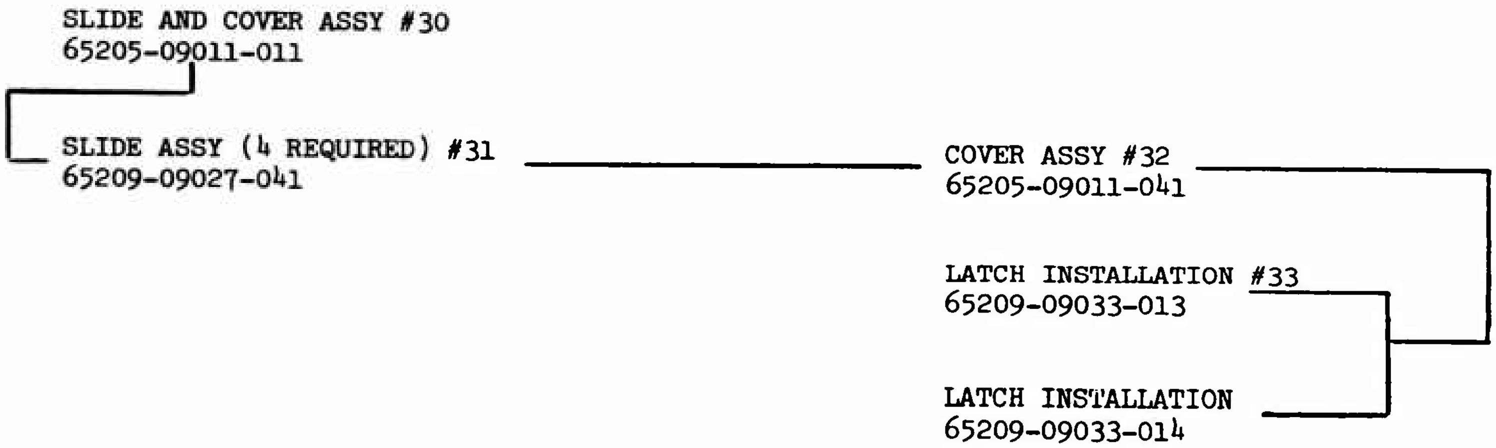
ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Pin, Retaining	21	65209-09032 -013, -014, -015, -016	2	Align cover for closing & hold in position	Bending	Inflight	Pin will not seat in retainer	Loss of structural integrity and increased aerodynamic loads on cover assy.	Damage to blades & fuselage. Increased drag.	Audio Report or unusual vibration inflight or visual insp. on landing	Immediate corrective action is not critical to A/C safety	One pin at each corner reduces the failure effect	Four pins required one at each corner of cover assy.
Cover Assy	22	65205-09010 -043	2	Aerodynamic fairing	Cracking	Inflight	Loss of structural integrity	None	Same as above	Same as above	Same as above	Aluminum stiffeners every 7½"	Fiber glass skin with aluminum stiffeners
					Delamination	SAME AS CRACKING							
Latch Installation	23	65209-09025 -011, -012	2	Secure cover to next section of M.R. Pylon	Distortion	Inspection & Maint.	Latch will not secure to catch	None	None	Visual Inspection		Redundant latches. Cover is hinged in "FAILSAFE" direction	
					Breaking	Inspection & Maint.	Latch will not secure or release	None	None	Visual Inspection			
Hinge Installation	24	65209-09023 -011, -012	2	Provide means of opening cover without removing from A/C	Breaking	Inspection & Maint.	Cover will not open or cover will not stay attached	None	None	Visual Inspection		Redundant hinges	
				Secure cover to M.R. pylon	Breaking	Inflight	Loss of securing function	SAME AS PIN, RETAINING - ABOVE					
Screen Installation	25	65209-09026 -011	2	Provide cooling air for APP & FOD prevention	Cracking	Inflight	Loss of structural integrity & protective function	None	None	Visual inspection on landing	Same as above		

TABLE XXVI. MAIN ROTOR PYLON FAIRING HINGE AND COVER ASSEMBLY RELIABILITY ANALYSIS

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO/ FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE		FAILURE MODE RATIO α	ENVIRONMENTAL RATIO K_E	OPERATIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/ HOUR OR CYCLE λ_G		FAILURE MODE CONTRIBUTION ($\alpha K_E K_A \lambda_G$)	COMPONENT CRITICALITY NO., C_r
Hinge & Cover Assy.	20	65205-09010 -011	2	Aerodynamic fairing	Cracking breaking distortion delamination misc.	All	Component - actual loss of function requiring repair or replacement	3-M Report 24 July '72 H/SH-3A/D -WUC 1122F,1123C 1123B,1123E How malf. SH-3A 9/67-10/68		.193 .182 .042 .050 .533	1.000 1.000 1.000 1.500 1.000	1.000 1.000 1.500 1.500 1.000	.0050 .0050 .0050 .0050 .0050		.0010 .0009 .0003 .0006 .0027	.0055
Pin, Retaining	21	65209-09032 -013, -014, -015, -016		Align cover for closing & hold in position	Bending	Inflight			HAZARD LEVEL II	Possible						
Cover Assy.	22	65205-09010 -043		Aerodynamic fairing	Cracking Delamination	Inflight Inflight			II II	Very Probable Probable	CRACKING OF ALUMINUM FRAMES LIKELY					
Latch Instl.	23	65209-09025 -011		Secure cover.	Distortion Breaking	Inspection & Maint. Inspection & Maint.			I I	Probable Possible						
Hinge Instl.	24	65209-09023 -011		Provide means of opening cover	Breaking	Inspection & Maint. Inflight			I	Possible						
Screen Instl.	25	65209-09026 -011		Cooling air inlet for APP	Cracking	Inflight			II	Probable						

TABLE XXVII. RELIABILITY LOGIC DIAGRAM - SLIDE AND COVER ASSEMBLY

Diagram No. 3



ALUMINUM STIFFENERS EVERY 8 INCHES

Areas of Possible Failure

1. Cracking of Aluminum Stiffeners
2. Breaking or Distortion of Latches
3. Delamination or Cracking of Skin

TABLE XXVIII. MAIN ROTOR PYLON FAIRING SLIDE AND COVER ASSEMBLY FAILURE MODE AND EFFECTS ANALYSIS

ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Slide Assy.	31	65209-09027-041	3	Provide means of opening cover without removing it from A/C	Distortion	Inspection & Maint.	Slide will jam on runner	Cover will not slide	None	Visual Inspection	Failure does not affect A/C safety	Each slide has two rollers	4 Slides required. All must function to slide cover.
					Breaking	SAME AS DISTORTION							
					Wear	SAME AS DISTORTION							
Cover Assy.	32	65205-09011-041	3	Aerodynamic fairing	Cracking	Inflight	Loss of structural integrity	None	Damage to blades & fuselage. Increased drag	Audio Report or unusual vibration in flight, or visual inspection on landing	Immediate corrective action is not critical to A/C safety	Aluminum stiffeners every 8".	Fiber glass skin with aluminum stiffeners
					Delamination	SAME AS CRACKING							
Latch Installation	33	65209-09033 -013, -014	3	Secure cover to next section of M.R. Pylon	Distortion	Inspection & Maint.	Latch will not secure to catch	None	None	Visual Inspection	Failure does not affect A/C safety	Redundant latches	
					Breaking	Inspection & Maint.	Latch will not secure or release	None	None	Visual Inspection	Failure does not affect A/C safety		

TABLE XXIX. MAIN ROTOR PYLON FAIRING SLIDE AND COVER ASSEMBLY RELIABILITY ANALYSIS

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO/FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE		FAILURE MODE RATIO α	ENVIRONMENTAL RATIO K_E	OPERATIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/HOUR λ_G		FAILURE MODE CONTRIBUTION $(\alpha K_E K_A \cdot \lambda_G)$	COMPONENT CRITICALITY NO., C_r
Slide & Cover Assy.	30	65205-09011-011	3	Aerodynamic	Cracking	All	Component - Actual loss of function requiring replacement or repair	3-M Report 24 July '72 H/SH-3A/S/G WUC 1122F, 1123Q, 1123B 1123E		.216	1.000	1.500	.0050		.0016	.0065
					Breaking					.171	1.000	1.500	.0050		.0013	
					Distortion					.027	1.500	1.500	.0050		.0003	
					Wear					.037	1.000	1.500	.0050		.0003	
					Delamination					.050	2.000	1.000	.0050		.0005	
					Misc.			How malf. SH-3A 9/67-9/68		.499	1.000	1.000	.0050		.0025	
Slide Assy.	31	65209-09027-041		Provide means of opening cover	Distortion	Inspection & Maint.			HAZARD LEVEL I	Probable						
					Breaking	Inspection & Maint.			I	Possible						
					Wear	Inspection & Maint.			I	Possible						
Cover Assy.	32	65205-09011-041		Aerodynamic fairing	Cracking	INFLIGHT			II	Probable	CRACKING WILL PROBABLY RESULT FROM DISTORTED SLIDE					
					Delamination	INFLIGHT			II	Probable						
Latch Installation	33	65209-09033-013, -014		Secure cover to M.R. Pylon	Distortion	Inspection & Maint.			I	Probable						
					Breaking	Inspection & Maint.			I	Possible						

TABLE XXX. RELIABILITY LOGIC DIAGRAM - NOSE GEAR DOOR

Diagram No. 4

DOOR INSTALLATION NOSE GEAR #40
65207-02007-011

DOOR ASSY, FWD #41
65207-02022-041

DOOR ASSY, AFT #42
65207-02024-042

PIN, HINGE #43
MS 20253P2-2860

FITTING ASSY, HINGE #44
65207-02010-041

FITTING ASSY (2) #45
65207-02008-041

Probable Areas of Failure

1. Cracking of Covers Where Fittings Attach
2. Delamination of Skin

TABLE XXXI. COCKPIT AND CANOPY DOOR INSTALLATIN, NOSE GEAR FAILURE MODE AND EFFECTS ANALYSIS

ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Door Assy. Fwd. Section	41	65207-02022-041	4	Aerodynamic Fairing	Cracking	INFLIGHT	Loss of structural integrity. Door may not close	Damage to nose gear tires or to aft section of door assy.	Damage to fuselage & blades. Increased aerodynamic drag.	Audio Report or unusual vibration in flight or visual inspection on landing	Immediate corrective action is not critical to A/C safety		Cracking most likely at fitting attachment points
					Delamination	INFLIGHT	SAME AS FOR CRACKING						
					Distortion	INFLIGHT	Door may not close	None	None	Same as above	Failure does not affect A/C safety		
Door Assy. Aft Section	42	65207-02024-042	4	Aerodynamic Fairing	Cracking	INFLIGHT	Loss of structural integrity, door may not close	Damage to nose gear tires or to fwd. section of Door Assy.	Damage to fuselage & blades. Increased aerodynamic drag.	Same as above	Immediate corrective action is not critical to A/C safety		Same as above
					Delamination	SAME AS CRACKING							
					Distortion	INFLIGHT	Door may not close	None	None	Same as above			
Pin, Hinge	43	MS20253P2-2860	4	Holds two sections of door together	Shear	INFLIGHT	Loss of aft floor section & damage to fwd. door section	None	Damage to blades & fuselage. Increased drag.	Same as above	Same as above	Continuous type hinge not likely to fail	
				Provides means of folding open door for ground clearance	Shear	Landing & Take-off	Damage to both door sections	None	None	Same as above	Failure does not affect A/C safety		

TABLE XXXI - Concluded

ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Fitting Assy. Hinge	44	65207-02010 -041	4	Provide means of opening door without removing it from A/C	Breaking	Landing & Takeoff	Loss of Hinge function	Damage to nose gear doors	None	Visual Inspection	Failure does not affect A/C safety	2 hinges prevent complete loss of doors	Loss of hinge may result in damage to doors
				Secure door to fuselage	Breaking	INFLIGHT	Loss of attachment function	Damage to nose gear doors	Damage to blades & fuselage. Increased drag.	Visual Inspection	Immediate corrective action not critical to A/C safety	Same as above	Same as above
Fitting Assy.	45	65207-02008 -041	4	Holds door in place	Breaking	All	Loss of structural integrity	Door will not be held in place	None	Visual Inspection	Failure does not affect A/C safety	2 fittings prevent complete loss of doors, doors are hinged in "failsafe" direction	

TABLE XXXII. COCKPIT AND CANOPY DOOR INSTALLATION, NOSE GEAR RELIABILITY ANALYSIS

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO/ FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE		FAILURE MODE RATIO α	ENVIRON-MENTAL RATIO K_E	OPERA-TIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/ HOUR λ_G		FAILURE MODE CON-TRIBUTION ($\alpha K_E K_A \lambda_G$)	COMPONENT CRITICALITY NO., C_r
Door Instl. Nose Gear	40	65207-02007-011	4	Aerodynamic Fairing	Cracking Breaking Delamination Distortion Misc.	All	Component - actual loss of function requiring repair or replacement	3-M Report 24 July '72 H/SH-3A/D/G WUC 1123 1123 How malf. SH-3A 9/67 - 10/63		.142 .205 .050 .061 .542	1.000 1.000 1.000 1.000 1.000	1.500 1.500 1.000 1.200 0.750	.0011 .0011 .0011 .0011 .0011		.0002 .0003 .0001 .0001 .0004	.0011
									Hazard Level							
Door Assy Forward	41	65207-02022-041		Aerodynamic Fairing	Cracking	INFLIGHT			II	Probable	CRACKING MOST LIKELY AROUND FITTING ATTACHMENT POINTS					
					Delamination	INFLIGHT			II	Possible						
					Distortion	INFLIGHT			II	Possible	DISTORTION MAY NOT AFFECT DOOR OPERATION					
Door Assy	42	65207-02024-042		Aerodynamic Fairing	Cracking	INFLIGHT			I	Probable	CRACKING MOST LIKELY AROUND FITTING ATTACHMENT POINTS					
					Delamination	INFLIGHT			I	Possible						
					Distortion	INFLIGHT			I	Possible	DISTORTION MAY NOT AFFECT DOOR OPERATION					
Pin, Hinge	43	MS20253P2-2869		Hold two sections of door together	Shear	Landing & Takeoff Inflight			II	None						
Fitting Assy Hinge	44	65207-02010-041		Provide means of opening door & securing it to fuselage	Breaking	Landing & Takeoff Inflight			II	Probable						

TABLE XXXIII. RELIABILITY LOGIC DIAGRAM - PERSONNEL DOOR

Diagram No. 4A

DOOR INSTALLATION, PERSONNEL #40A
65207-03006-011

DOOR ASSY, LOWER #41A 65207-03018-041	SUPPORT INSTALLATION #42A 65207-03013-011	LATCH INSTALLATION #45A 65207-03019-011
SUPPORT ASSY #43A 65207-03014-081	CABLE ASSY #44A 65207-03013-083	HINGE (2 REQUIRED) #46A P/N UNK
SUPPORT ASSY 65207-03014-083	CABLE ASSY 65207-03013-084	

DUAL PIN LATCH
REDUNDANT SUPPORT ASSY

Probable Areas of Failure

1. Cracks or Punctures in Door and Steps, Deformed Steps
2. Breaking Hinge and Support Fittings
3. Distortion of Latch

Failure of Latch Likely Due to Complex Mechanism

TABLE XXXIV. FUSELAGE DOOR INSTALLATION, PERSONNEL FAILURE MODE AND EFFECTS ANALYSIS

ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Door Assy Lower	41A	65207-03018 -041	4A	Personnel Access to Aircraft	Cracking of steps or door skin at step attachment	Ground	Loss of structural integrity	None	None	Visual Inspection	Failure does not affect A/C		Possible injury to personnel
					Distortion	Ground	Loss of structural integrity	Door may not close	None	Visual Inspection	Same as above	Redundant support installation	
					Dents or bending of steps	Ground	Loss of structural integrity	None	None	Visual Inspection	Same as above		Rough usage likely in this area
				Aerodynamic Fairing	Cracking of exterior skin	INFLIGHT	Loss of structural integrity	Damage to upper door	Damage to blades & fuselage Increased Drag	Visual Inspection	Immediate corrective action is not critical to A/C safety		
Support Assy	43A	65207-03014 -081, -083	4A	Support lower door in Airstair configuration	Wear at cable or door attachments	Ground	Loss of support function	Damage to lower door	None	Visual Inspection	Failure does not affect A/C safety	Redundant support installation	Excessive lateral forces likely
					Cracking of support	SAME AS WEAR							
Cable Assy	44A	65207-03013 -083, -084	4A	Support lower door in Airstair configuration	Breaking of individual strands	Ground	Loss of support function	Damage to lower door	None	Visual Inspection	Same as above	Multiple stranded cable & redundant support installation	

TABLE XXXIV - Concluded

ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Latch Instl.	45A	65207-03019	4A	Secure door in closed position	Distortion	Before & after flight	Loss of securing function	Lower door will not secure	None	Visual Inspection	Failure does not affect A/C safety		Rough usage likely in this area
					Wear	SAME AS DISTORTION							
				Breaking	Before & after flight	SAME AS DISTORTION							
				Breaking	INFLIGHT	Loss of securing function	Loss of, or damage to upper & lower doors	Damage to fuselage & blades. Increased drag	Audio report or visual inspection	Audio report or visual inspection	Immediate corrective action is not critical to A/C safety	Dual Pin latch may prevent complete loss of door	
Hinge	46A	UNK	4A	Provide means of opening door without removing it from A/C	Cracking	Before & after flight	Loss of structural integrity	Damage to lower door	None	Visual Inspection	Failure does not affect A/C safety		Rough usage likely in this area. 2 hinges required for proper operation

TABLE XXXV. FUSELAGE DOOR INSTALLATION, PERSONNEL RELIABILITY ANALYSIS

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO/ FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE		FAILURE MODE RATIO α	ENVIRONMENTAL RATIO K_E	OPERATIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/HOUR λ_G		FAILURE MODE CONTRIBUTION ($\alpha K_E K_A \cdot \lambda_G$)	COMPONENT CRITICALITY NO., C_r
Door Instl. Personnel	40A	65207-03006-011	4A	Personnel access to aircraft	Cracking Breaking Distortion Wear Misc.	All	Component - actual loss of function resulting in repair or replacement	3-M Report 24 July '72 H/SH-3A/D/G WUC 11227, 11228, 1122A		.109 .204 .037 .103 .547	1.000 1.000 1.000 1.000 1.000	1.000 1.000 1.000 1.000 1.000	.0186 .0186 .0186 .0186 .0186		.0020 .0038 .0007 .0019 .0102	.0186
Door Assy Lower	41A	65207-03018-041		Personnel access to aircraft & Aerodynamic Fairing	Cracking Distortion Dents	Grnd. &Flt. Ground Ground			Hazard Level I I I	Probable Possible Probable						
Support Assy	43A	65207-03014-081, -083		Support lower door in Airstair configuration	Wear Cracking	Ground Ground			I I	Probable Possible						
Cable Assy	44A	65207-03013-083, -084		Support lower door in Airstair configuration	Breaking of individual strands	Ground			I	Probable						

TABLE XXXV - Concluded

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO. FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE		FAILURE MODE RATIO α	ENVIRONMENTAL RATIO K_E	OPERATIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/HOUR λ_G		FAILURE MODE CONTRIBUTION $(\alpha K_E K_A \cdot \lambda_G)$	COMPONENT CRITICALITY NO., C_r
								Hazard Level								
Latch Instl	45A	65207-03019-011	4A	Secure door in closed position	Distortion	Pre & Post Flight		I	Possible							
					Wear	Same as above		I	Possible							
					Breaking	Pre & Post Flight, in flight		II	Not very possible							
Hinge	46A	UNK		Secure door to fuselage	Cracking	Pre & Post Flight in flight		II	Possible							

TABLE XXXVI. RELIABILITY LOGIC DIAGRAM - FUEL CELL COVER

Diagram No. 5

COVER INSTALLATION, FUEL CELL #50
65207-08005-011/-012

COVER ASSY #51
65207-08005-041/-042

PLUG, DIMPLED #52
SL601-3-6C

Probable Areas of Failure

1. Cracking or Delamination of Cover
2. Puncture of Cover or Plug From Rough Use

TABLE XXXVII. SPONSON COVER INSTALLATION, FUEL CELL FAILURE MODE AND EFFECTS ANALYSIS

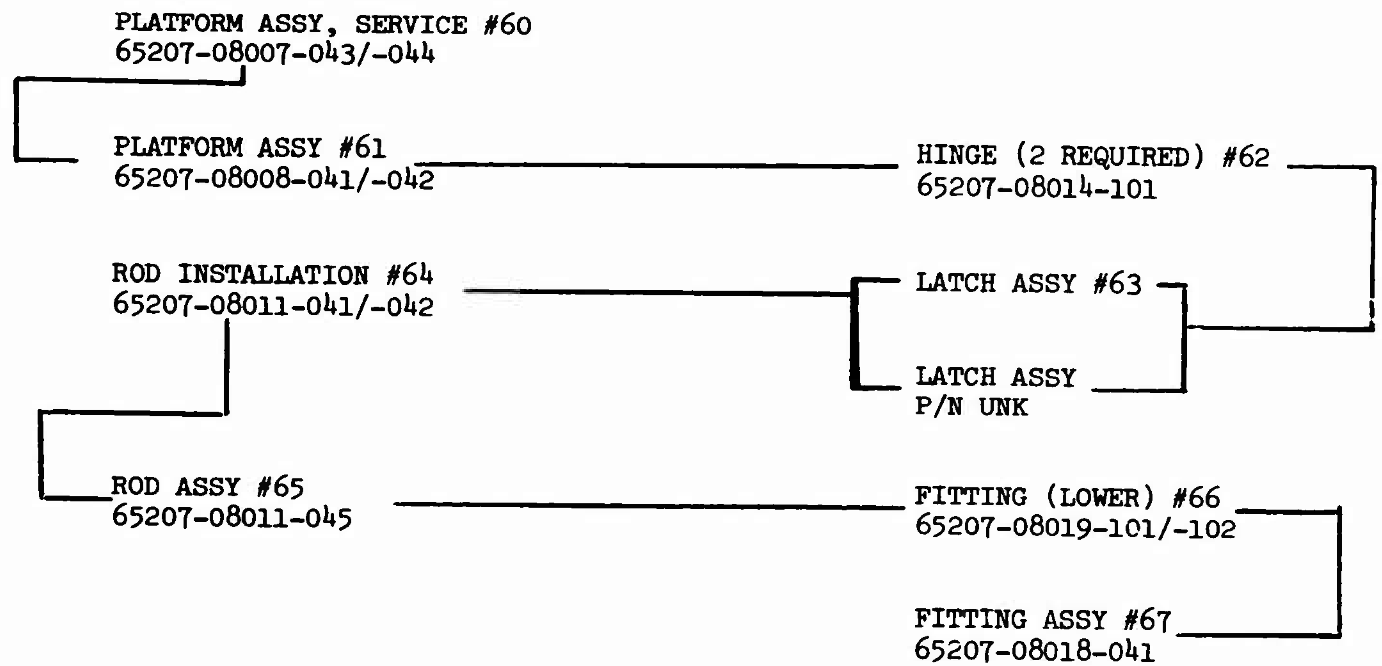
ITEM IDENTIFICATION				FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECT ON			FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER				COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM				
Cover Assy	51	65207-08005 -041, -042	5	Work Platform	Puncture	Inspection & Maint.	Loss of structural integrity	Loss of fuel cell protection	None	Visual Inspection	Failure does not affect A/C safety	Extra plies at attachment points. Attachment screws every 3" along all edges	Rough usage likely in this area
					Cracking	SAME AS PUNCTURE							
					Delamination	SAME AS PUNCTURE							
				Aerodynamic	Cracking	INFLIGHT	Loss of structural integrity	None	Increased Aerodynamic Drag	Visual Inspection	Immediate corrective action is not critical to A/C safety	Same as above	
					Delamination	SAME AS CRACKING							
Plug, Dimpled	52	SL601-3-6C	5	Fuel Cell Access	Puncture	Inspection & Maint.	Loss of structural integrity	Loss of Fuel Cell protection	None	Visual Inspection	Failure does not affect A/C safety	Attachment screws every 3½" around circumference	
					Cracking	SAME AS PUNCTURE							
				Aerodynamic Fairing	Cracking	INFLIGHT	Loss of structural integrity	None	Increased Drag	Visual Inspection	Immediate corrective action not critical to A/C safety	Same as	

TABLE XXXVIII. SPONSON COVER INSTALLATION, FUEL CELL RELIABILITY ANALYSIS

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO/ FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE		FAILURE MODE RATIO α	ENVIRON-MENTAL RATIO K_E	OPERA-TIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/ HOUR λ_G		FAILURE MODE CON-TRIBUTION $(\alpha K_E K_A \cdot \lambda_G)$	COMPONENT CRITICALITY NO., C_r
Cover Instl.	50	65207-	5	Work Plat-form & Aerodynamic Fairing	Cracking Puncture Delamination	All	Component - actual loss of function requiring repair or replacement	3-M Report 24 July '72 H/SH-3A/D/G WUC 11238, 1123T How Malf. SH-3A 9/67 - 10/68		.347 .050 .050 .553	1.500 1.000 1.000 1.500	1.000 1.000 1.000 1.000	.0011 .0011 .0011 .0011		.0006 .0001 .0001 .0009	.0017
									Hazard Level							
Cover Assy	51	65207-08005-041, -042		Work Plat-form & Aerodynamic Fairing	Puncture	Inspection & Maint.			I	Possible						
					Cracking	Inspection & Maint. INFLIGHT			II	Probable						
					Delamination	Inspection & Maint. INFLIGHT			II	Possible						
Plug, Dimpled	52	SL601-3-6C		Fuel Cell Access & Aerodynamic Fairing	Puncture	Inspection & Maint.			I	Not very possible						
					Cracking	Inspection & Maint. INFLIGHT			II	Probable						

TABLE XXXIX. RELIABILITY LOGIC DIAGRAM - SERVICE PLATFORM

Diagram No. 6



PLATFORM IS HINGED IN "FAIL SAFE" DIRECTION
REDUNDANT LATCHES. HINGE SPRING LOADED CLOSED

Probable Areas of Failure

1. Punctures and Cracks at Fittings From Rough Use

TABLE XL. SPONSON PLATFORM ASSEMBLY, SERVICE FAILURE MODE AND EFFECTS ANALYSIS

ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Platform Assy	61	65207-08008 -041, -042	6	Work Platform	Puncture	Inspection & Maint.	Loss of structural integrity	None	None	Visual Inspection	Failure does not affect A/C safety		Rough usage by Maint. personnel is likely Cracking around support fittings likely
					Cracking	SAME AS PUNCTURE							
					Delamination	SAME AS PUNCTURE							
				Aerodynamic Fairing	Delamination	INFLIGHT	Loss of structural integrity	None	Damage to blades & fuselage. Increased Drag	Visual Inspection	Immediate corrective action is not critical to A/C safety		
					Cracking	SAME AS DELAMINATION							
Hinge	62	65207-08014	6	Provide means of opening platform for service function	Breaking	Maint. & Inspection	Loss of hinge function	Damage to platform assy	None	Visual Inspecton	Failure does not affect A/C safety	2 hinges prevent complete loss of platform	Loss of 1 hinge may result in damage to platform. Hinge spring loaded closed
				Secure platform to fuselage	Breaking	INFLIGHT	Loss of attachment function	Damage to platform assy	Damage to blades & fuselage. Increased Drag	Visual Inspection	Immediate corrective action is not critical to A/C safety	Same as above	
Latch Assy	63	UNK	6	Secure platform to fuselage	Distortion	Maint. & Inspection	Latch will not secure to catch	None	None	Visual Inspection		Redundant latches. Platform hinged in "failsafe" direction	
					Breaking	SAME AS DISTORTION							

TABLE XL - Concluded

ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Rod Assy	65	65207-08011-045	6	Support platform during Inspec.	Breaking	Inspection & Maint.	Loss of support function	Damage to platform & sponson assy	None	Visual Inspection	Failure does not affect A/C safety		Possible injury to maintenance personnel
Fitting	66	65207-08019-	6	Attach Rod assy to fuselage	Breaking	Inspection & Maint.	Loss of structural integrity	Damage to platform & sponson assy	None	Visual Inspection	Same as above		Same as above
Fitting Assy	67	65207-08018-041	6	Attach Rod assy to fuselage	Breaking	Inspection & Maint.	Loss of structural integrity	Damage to platform & sponson assy	None	Visual Inspection	Same as above		Same as above

TABLE XLI. SPONSON PLATFORM ASSEMBLY, SERVICE RELIABILITY ANALYSIS

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO/ FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE		FAILURE MODE RATIO α	ENVIRONMENTAL RATIO K_E	OPERATIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/ HOUR λ_G		FAILURE MODE CONTRIBUTION $(\alpha K_E K_A \cdot \lambda_G)$	COMPONENT CRITICALITY NO., C_r
Platform Assy, Service	60	65207-08008-041, -042	6	Work Platform & Aerodynamic Fairing	Cracking	All	Component - actual loss of function resulting in repair or replacement	3-M Report 24 July '72 H/SH-3A/D/G WUC 11236, 11238, 1123D 1123E How malf. SH-3A 9/67 - 10/68		.159	0.800	1.000	.0066		.0008	.0054
					Breaking					.150	0.800	1.000	.0066		.0008	
					Puncture					.050	1.000	1.000	.0066		.0008	
					Delamination					.050	1.000	1.000	.0066		.0003	
					Distortion					.051	0.800	1.000	.0066		.0003	
					Misc.					.540	0.800	1.000	.0066		.0029	
									Hazard Level							
Platform Assy	61	65207-08008-041, -042		Work Platform & Aerodynamic Fairing	Puncture	Inspection & Maint.			I	Probable						
					Cracking	Inspection & Maint. INFLIGHT			II	Probable						
					Delamination	Inspection & Maint. INFLIGHT			II	Probable						
Hinge	62	65207-08014-101		Secure Platform to fuselage	Breaking	Inspection & Maint. INFLIGHT			II	Not very possible						
Latch Assy	63	UNK		Secure Platform to fuselage	Distortion	Maint. & Inspection			I	Probable						
					Breaking	Maint. & Inspection			I	Possible						

TABLE XLI - Concluded

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO./FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE	Hazard Level	FAILURE MODE RATIO α	ENVIRONMENTAL RATIO K_E	OPERATIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/HOUR λ_G		FAILURE MODE CONTRIBUTION $(\alpha K_E K_A \cdot \lambda_G)$	COMPONENT CRITICALITY NO., C_r
Rod Assy	65	65207-08011-045	6	Support platform during insp. & maint.	Breaking	Inspection & Maint.			I	None						
Fitting	66	65207-08019-101, -102		Attach platform to rod	Breaking	Inspection & Maint.			I	None						
Fitting	67	65207-08018-041		Attach rod to fuselage	Breaking	Inspection & Maint.			I	None						

TABLE XLII. RELIABILITY LOGIC DIAGRAM - WORK PLATFORM

Diagram No. 7

PLATFORM ASSY WORK #70
65207-09004-041/-042

PANEL ASSY, UPPER #71
65207-09004-083/-084

PANEL ASSY, LOWER #72
65207-09004-081/-082

HINGE HALF (4 REQUIRED) #74
65207-09004-133/-134/-135/-138

LATCH ASSY, #73
65207-09004-043

LATCH ASSY
65207-09004-043

PIN, HINGE (2 REQUIRED) #75
65207-09004-113

CONTINUOUS TYPE HINGE AND PIN
REDUNDANT LATCHES

Probable Areas of Failure

1. Punctures or Cracks From Rough Use
2. Distorted Platforms
3. Working Loose of Pins or Screws

TABLE XLIII. MAIN ROTOR PYLON FAIRING PLATFORM ASSEMBLY, WORK FAILURE MODE AND EFFECTS ANALYSIS

ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Panel Assy Upper	71	65207-09004 -083, -084	7	Platform Support	Cracking	Maint. & Inspection	Loss of Support	Loss of structural integrity & platform safety	None	Visual Inspection	Failure does not affect A/C safety		Rough usage of upper & lower Panel likely
					Delamination	SAME AS CRACKING							
					Distorion	Maint. & Inspection	Panel will not secure in support position	Platform will not close	None	Visual Inspection	Same as above		
				Aerodynamic Fairing	Cracking	INFLIGHT	Loss of structural integrity	Damage to platform components	Damage to blades & fuselage. Increased drag	Audio report or unusual vibration, or visual inspection on landing	Immediate corrective action is not critical to A/C safety		
					Delamination	SAME AS CRACKING							
Panel Assy Lower	72	65207-09004 -081, -082	7	Work Platform	Cracking	Maint. & Inspection	Loss of structural integrity	Loss of structural integrity & platform safety	None	Visual Inspection	Failure does not affect A/C safety		Same as above
					Delamination	SAME AS CRACKING							
					Distortion	Maint. & Inspection	Panel will not secure in work position	Platform will not close	None	Visual Inspection	Same as above		

TABLE XLIII - Concluded

ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Panel Assy Lower (cont.)	72	65207-09004 -081, -082	7	Aerodynamic Fairing	Cracking	INFLIGHT	Loss of structural integrity	Damage to platform components	Damage to blades & fuselage. Increased drag	Audio report or unusual vibration, or visual inspection on landing	Immediate corrective action is not critical to A/C		
					Delamination	SAME AS CRACKING							
Latch Assy	73	65207-09004	7	Secure platform to fuselage	Breaking	Maint. & Inspection	Latch will not secure or release platform from catch	None	None	Visual Inspection	Failure does not affect A/C safety	Redundant latches	Failure probably due to rough usage
Hinge, Half	74	65207-09004 -133, -134, -135, -138,	7	Provide means of folding platform into work configuration	Breaking	Maint. & Inspection	Loss of folding capability	Damage to platform panels & loss of work function	None	Visual Inspection	Same as above	Continuous hinge & pin	hinge halves required for proper operation
Pin, Hinge	75	65207-09004 -113	7	Secure hinge halves together	Breaking	Maint. & Inspection	Loss of folding capability	None	None	Visual Inspection	Same as above	Same as above	2 pins required for proper operation
				Secure platform halve together & to fuselage	Working loose	INFLIGHT	Loss of Pin	Loss of, or damage to platform	Damage to blades & fuselage	Audio report or unusual vibration, or visual inspection on landing	Immediate corrective action is not critical to A/C safety		

TABLE XLIV. MAIN ROTOR PYLON FAIRING PLATFORM ASSEMBLY, WORK RELIABILITY ANALYSIS

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO/FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE	Hazard Level	FAILURE MODE RATIO α	ENVIRONMENTAL RATIO K_E	OPERATIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/HOUR λ_G		FAILURE MODE CONTRIBUTION ($\alpha K_E K_A \lambda_G$)	COMPONENT CRITICALITY NO., C_r
Platform Assy, Work	70	65207-09004-041, -042	7	Work Platform & Aerodynamic Fairing	Cracking Breaking Distortion Delamination Misc.	All	Component-actual loss of function requiring replacement or repair	3-M Report 24 July'72 H/SH-3A/D/G WUC 11236 1123B, 1123D 1123E How malf. SH-3A 9/67 - 10/68		.159 .150 .051 .050 .590	0.800 0.800 0.800 1.000 0.800	.0066 .0066 .0066 .0066 .0066			.0008 .0008 .0003 .0003 .0031	.0053
Panel Assy Upper	71	65207-09004-083, -084		Platform Support & Aerodynamic Fairing	Cracking	Maint. & Inspection, in flight			II	Probable						
					Delamination				II	Possible						
					Distortion				II	Not very possible						
Panel Assy Lower	72	65207-09004-081, -082		Work Platform & Aerodynamic Fairing	Cracking	Maint. & Inspection, in flight			II	Probable						
					Delamination				II	Possible						
					Distortion				II	Not very possible						
Latch Assy	73	65207-09004-043		Secure platform to fuselage	Breaking	Maint. & Inspection			I	Probable						

TABLE XLIV - Concluded

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO./FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE	Hazard Level	FAILURE MODE RATIO α	ENVIRONMENTAL RATIO K_E	OPERATIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/HOUR λ_G		FAILURE MODE CONTRIBUTION ($\alpha K_E K_A \cdot \lambda_G$)	COMPONENT CRITICALITY NO., C_r
Hinge Half	74	65207-09004-133, -134, -135 -138	7	Provide means of folding platform into work configuration	Breaking	Maint. & Inspection			I	Possible						
Pin, Hinge	75	65207-09004-113		Secure hinge halves	Breaking	Maint. & Inspection			I	Not very possible						
					Working loose	INFLIGHT			II	Not very possible						

TABLE XLV. RELIABILITY LOGIC DIAGRAM - EAPS REAR FRAME

Diagram No. 8

REAR FRAME ASSY, EAPS #80
65302-12503-041

REAR FRAME #81
65302-12514-101

REAR FLANGE #82
65302-12515-101

LATCH #83
H 738-1

LATCH
H 738-1

Probable Areas for Failure

1. Cracking at Rivet Holes
2. Breaking of Flange During Maintenance and Inspection

TABLE XLVI. EAPS, REAR FRAME ASSEMBLY FAILURE MODE AND EFFECTS ANALYSIS

ITEM IDENTIFICATION							FAILURE EFFECT ON						
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER	FUNCTION	FAILURE MODE	OPERATION PHASE	COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM	FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
Rear Frame	81	65302-12514 -101	8	Provides support for rear flange	Cracking	INFLIGHT	Loss of support function	Damage to EAPS	None	Visual inspection on landing	Immediate corrective action is not critical to A/C safety	EAPS hinged in "FAILSAFE" direction	Aluminum construction cracking from vibration likely
Rear Flange	82	65302-12515 -101	8	Provides support for latches & fairing for EAPS - Engine interface	Cracking	INFLIGHT	Loss of structural integrity	Damage to engine bellmouth (ingestion not likely)	None	Visual Inspection	Same as above	Same as above	Same as Above
					Distortion	Inspection & Maint.	Loss of fairing function	EAPS will not align in front of engine	None	Visual Inspection	Failure does not affect A/C safety		Bending likely during Maint. & Inspection
Latch	83	H 738-1	8	Attach EAPS to engine	Cracking	INFLIGHT	Loss of structural integrity	None	None	Visual Inspection on landing	Immediate corrective action is not critical to A/C safety	Same as above & redundant latches	
					Distortion	Inspection & Maint.	Latch will not secure to catch	None	None	Visual Inspection	Failure does not affect A/C safety		

TABLE XLVII. EAPS, REAR FRAME ASSEMBLY RELIABILITY ANALYSIS

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO/ FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE		FAILURE MODE RATIO α	ENVIRONMENTAL RATIO K_E	OPERATIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/ HOUR λ_G		FAILURE MODE CONTRIBUTION ($\alpha K_E K_A \cdot \lambda_G$)	COMPONENT CRITICALITY NO., C_r
Rear Frame Assy	80	65302-12503-041	8	Support EAPS	Cracking	All	Component - actual loss of function resulting in repair or replacement	3-M Report 24 July'72 H/SH-3A/D/G WUC 11217, 11238, 1123E How malf. SH-3A 9/67 - 9/68		.145 .049 .806	1,000 1,000 1,000	1.000 1.000 1.000	.0017 .0017 .0017		.0003 .0001 .0011	.0015
Rear Frame	81	65302-12514-101		Provides support for rear flange	Cracking	INFLIGHT			Hazard Level II	Probable						
Rear Flange	82	65302-12515-101		EAPS - Eng. Interface	Cracking Distortion	INFLIGHT Inspection & Maint.			II I	Probable Possible						
Latch	83	H 738-1		Attach EAPS to Engine	Cracking Distortion	INFLIGHT Inspection & Maint.			II I	Probable Probable						

TABLE XLVIII. RELIABILITY LOGIC DIAGRAM - COMPASS TRANSMITTER SUPPORT

Diagram No. 9

SUPPORT INSTALLATION, COMPASS TRANSMITTER #90
6420-65143-011

SUPPORT ASSY #91
6420-65143-041

Probable Areas of Failure

1. Cracking at Rivet Holes and Bends
2. Bending and Dents From Maintenance Personnel

TABLE XLIX. TAIL BOOM SUPPORT INSTALLATION, COMPASS TRANSMITTER FAILURE MODE AND EFFECTS ANALYSIS

ITEM IDENTIFICATION				FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECT ON			FAILURE DETECTION METHOD	CORRECTIVE ACTION TIME AVAILABLE/ TIME REQUIRED	DESIGN PROVISIONS TO REDUCE CRITICALITY	REMARKS
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NUMBER				COMPONENT/ FUNCTIONAL ASSEMBLY	NEXT HIGHER SUBSYSTEM	UPPERMOST SYSTEM				
Support Assy	91	6420-65143-041	9	Secure compass transmitter in tail boom	Cracking at rivet holes & at bends	All	Loss of structural integrity	Damage to transmitter	None	Loss of compass indication or visual inspection	Failure does not affect A/C safety		
					Bending from Maint. Personnel	All	SAME AS CRACKING						

TABLE L. TAIL BOOM SUPPORT INSTALLATION RELIABILITY ANALYSIS

ITEM IDENTIFICATION				FAILURES				RELIABILITY EVALUATION								
NAME	IDENT. NO.	DRAWING REFERENCE DESIGNATION	RELIABILITY LOGIC DIAGRAM NO/ FUNCTION NO.	FUNCTION	FAILURE MODE	OPERATION PHASE	FAILURE EFFECTS	RELIABILITY DATA SOURCE CODE		FAILURE MODE RATIO α	ENVIRON-MENTAL RATIO K_E	OPERA-TIONAL RATIO K_A	GENERIC FAILURE RATE FAILURES/ HOUR λ_G		FAILURE MODE CON-TRIBUTION ($\alpha K_E K_A \cdot \lambda_G$)	COMPONENT CRITICALITY NO., C_r
Support Installation	90	6420-65143-011	9	Support, Compass Transmitter	Cracking Bending Misc	All	Component-actual loss of function requiring repair or replacement	3-M Report 24 July '72 H/SH-3A/D/G WUC 11217 How Malf. SH-3A 9/67 - 10/68		.375 .050 .575	1.000 1.200 1.000	1.000 1.200 1.000	.0003 .0003 .0003		.0001 .0001 .0002	.0004
Support Assy	91	6420-65143-041		Support Compass Transmitter	Cracking Bending	All All			Hazard Level I I	Very Probable Probable						

APPENDIX III

EXPERIMENTAL TEST PROCEDURE

ETP 6598-11933

TITLE: Helicopter Secondary Structures
Structural Endurance Test of

DATE: February 20, 1973

1.0 SCOPE

1.1 Purpose - The purposes of these tests are:

(1) Duplicate failure modes experienced in service with the present H-53 designs of:

(a) The main rotor pylon hinge and cover assembly.
(b) The lower personnel door assembly.
(c) The main rotor pylon work platform assembly.

(2) Demonstrate increased adequacy of the redesigned versions of the above components.

1.2 Background - These tests are part of a U. S. Army research contract concerning the maintenance problems and aircraft down time resulting from problems experienced on secondary structure components. Included in this study are the following:

(1) Identifying the three secondary structures accounting for the highest maintenance man-hours and/or aircraft down time.

(2) Determining the inadequacies of present design and/or test criteria presently used for designing and qualifying secondary structures.

(3) Recommendations for addition and/or revision to existing design and test specifications.

(4) Redesign of the three selected structures to new and/or revised criteria.

(5) Test of the original and redesigned structures to:

(a) duplicate the in-service inadequacies of the original designs and
(b) demonstrate improvement of the redesigned structure.

Problems have been experienced in-service with various secondary structure

assemblies. It is felt that some of these problems stem from deficiencies of present design and qualification criteria that do not adequately specify in-service usage and environment. Definition of new and revised criteria is being considered as a means of increasing secondary structure reliability.

A review of operational and overhaul maintenance data for the Marine CH-53 A/D, Air Force HH-53, and Army CH-54A helicopters showed that three secondary structure assemblies (the main rotor pylon hinge and cover assembly, the lower personnel door, and the main rotor pylon work platform assembly), shown in Figure 1, account for high maintenance man-hours and aircraft down time. These tests are designed to produce field failure modes of these components in the lab and to demonstrate improved performance of the redesigned assemblies. This test plan is presented to meet the test plan requirements of the contract, Reference 2.1.

2.0 APPLICABLE DOCUMENTS

2.1 Contract No. DAAJ02-72-C-0070

2.2 Sikorsky Aircraft Drawing 65205-09010, "Hinge and Cover Assembly, Main Rotor Pylon."

2.3 Sikorsky Aircraft Drawing 65207-03018, "Personnel Door Assembly."

2.4 Sikorsky Aircraft Drawing 65207-09004, "Work Platform Installation, Main Rotor Pylon."

2.5 Sikorsky Engineering Letter SEL-2209, dated 19 October 1972, "Helicopter Secondary Structures Reliability and Maintainability Investigation Progress Report."

2.6 Sikorsky Aircraft Drawing SL65M-01162, "Secondary Structures Endurance Test."

2.7 Sikorsky Aircraft Drawing SL65M-01164, "Secondary Structures Endurance Test."

2.8 Sikorsky Aircraft Drawing SL65M-01165, "Secondary Structures Endurance Test."

3.0 REQUIREMENTS

3.1 Experimental Test Design

3.1.1 General - The problem areas of the three selected components have been reviewed to determine the service conditions (loading spectra, vibration and aerodynamic environment, and abuse) which most likely contributed to the in-service failure modes of each part. Laboratory tests have been designed to integrate these conditions and the revised testing requirements (presented in Appendix (I) of this final report) into a combined test spectra designed to duplicate these in-service modes in the laboratory.

Scheduled usage and estimates of nonscheduled usage and abuse were utilized to combine the individual service conditions into composite test programs which allow the interaction of these conditions in proportion to field exposure. No attempt shall be made to quantify the reliability of present or redesigned parts with these spectra; however, comparisons utilizing the spectra of present and redesigned versions and/or estimate of equivalent service life shall be possible.

It is assumed that in the course of these tests, the present design components will experience the in-service failure modes and that the redesigned components will demonstrate improved performance. Since there are several different modes of failure associated with each of the selected components and since only one specimen each of present and redesigned configuration will be tested, the following decisioning procedure shall be followed when failures are encountered:

(1) Provisions shall be made for repair/replacement of those failure modes which have not previously been experienced in-service on present design assemblies.

(2) Should an in-service mode of failure be experienced, testing shall continue with the failed part if:

   (a) Continuing operation with the failed part is considered to be a contributing factor in generating other in-service modes, or
   (b) Usage feedback from the field indicates that such operation is possible.

(3) Should a new failure mode be experienced on the redesigned assemblies, testing shall continue with the failed part if (2) (b) above is satisfied.

   Note that during the course of testing, normal maintenance and adjustment procedures for each assembly shall be followed at the prescribed intervals.

   Testing of each assembly shall continue until either all field failure modes are duplicated or 500 hours of equivalent flight time have elapsed.

3.1.2 Main Rotor Pylon Hinged Cover Assembly - The main rotor pylon hinged cover assembly P/N 65205-09010 has experienced the following problems in service:

(1) Cracking and delaminating, distorting and breaking of the fiber glass cover.

(2) Distorting and breaking of the hinges and the retention latch installation. These problems have resulted from the combination of aerodynamic loading, vibration, normal usage, and abuse of the hinged cover.

One production cover assembly shall be subjected to laboratory tests designed to duplicate problem modes experienced in the field. Testing shall include the combined effects of water and salt spary, vibration and aerodynamic loading, as well as normal and abusive cycling of the cover. The water and salt spray shall simulate high humidity and salt air environment which results in possible hardware corrosion.

3.1.2.1 Vibration Testing - The vibration and aerodynamic loading of the cover shall be simulated using an eccentric mass driven by a vari-drive motor attached to the cover in such a manner as to open the cover from its latched position. The frequency and magnitude of the applied loading shall be determined from in-flight measurements of cover vibrations which shall be performed in conjunction with an existing flight test program.

3.1.2.2 Cycling - Normal and abusive cycling of the cover shall include manual cycling from the latched to the fully opened position as well as free-fall drops to the open and closed positions and slamming of the cover. The frequency of the open/close cycling shall be 3/flight hour as determined from a review of scheduled inspections and unscheduled maintenance which require opening of the cover. Since no statistics are kept on abusive use of the cover, it is assumed that one cycle per hour shall be an abusive cycle consisting of free-fall to the open position and free-fall or slam to the latched position.

NOTE: The first five abusive cycles shall consist of free-falls to the open and closed positions from a fully vertical open position. Thereafter, an abusive cycle shall consist of a drop or slam from a height of 18 inches to the fully closed and open positions.

3.1.2.3 Procedure - Testing shall be conducted in programmed blocks designed to allow interaction of the simulated aircraft vibrations and the open/close cycling, yet minimize the number of test setup changes. A typical test block (see Section 4.1) shall consist of 20 hours of vibration testing and 60 manual open/close cycles, 20 of which shall consist of free-fall to the open position and free-fall or slam to the latched position. Functional checks of the cover, latches, and hinges shall be performed prior and subsequent to each test block. In addition, periodic visual inspections shall be performed to check for distorting and release of the retention latch mechanism, distorting or breaking of the hinges, and distorting, delaminating or breaking of the fiber glass structure. Normal maintenance shall be performed at the prescribed intervals. Should breaking or failure of any cover assembly component occur, the provisions of Section 3.1.1 shall apply. Testing shall be repeated on the redesigned hinged cover assembly and the results of such testing analyzed to determine in-service improvement.

3.1.3 Lower Personnel Door Assembly - The lower personnel door assembly P/N 65207-03018-041 (-03006-011 installation) has experienced the following problems in service:

(1) Breaking of the door support cables.

(2) Weakening and cracking of the hinge and support assembly.

(3) Distortion, wear, and breaking of the latch installation.

(4) Cracking of the center stair riser and exterior skin of the door.

These problems have resulted from the combination of vibration, normal usage, and abuse of the lower personnel door.

The following paragraphs outline the laboratory tests designed to subject one production lower personnel door to simulated field problem modes.

3.1.3.1 Cycling - The lower personnel door shall be cycled between its deployed and stowed positions to simulate the abusive handling of the door encountered in the field resulting in broken cables, worn cable bracket holes, and crack initiation due to structural shock. Each cycle shall consist of raising the door from its deployed position, slamming closed, latching and unlatching, and permitting the door to free-fall back to its deployed position. This procedure shall be repeated for a total of eight complete cycles based on three entries/exits from the aircraft for each 1½ hours of scheduled flight.

3.1.3.2 Vibration - The lower personnel door, in its stowed and latched configuration, shall be vibrated and aerodynamically loaded consistent with actual flight inputs. The frequency and magnitude of the applied loading shall be determined from in-flight measurements of door vibrations encountered during a flight test program. The vibration shall be simulated using a vari-drive motor and eccentric mass. The door shall be vibrated for a period of eight hours.

3.1.3.3 Stair Tread Impact - The lower personnel door, while in its deployed configuration, shall receive an impact load (400 lb) from a weighted Government-issue combat boot on and perpendicular to the bottom stair tread, simulating loads imposed by personnel jumping on the step during entry into and exit from the aircraft. The force and acceleration of such loading shall be derived from human factors information available. This loading shall be repeated for a total of 90 impacts based on estimated average personnel flight capacities, scheduled inspections, and unscheduled maintenances.

3.1.3.4 Stair Riser Impact - The lower personnel door, while in its deployed configuration, shall receive an impact load (250 lb) from a weighted Government-issue combat boot on and perpendicular to the center stair support riser, simulating loads imposed by personnel kicking the riser on rapid entry into the aircraft. This abuse has resulted in cracked center stair risers. The force and acceleration of such loading shall be derived from human factors information available. This loading shall be repeated for a total of 40 impacts based on estimated average personnel impacts per four hours of flight time.

3.1.3.5 Support Cable Impact - With the lower personnel door in its de-

ployed configuration, an impact load (400 lb) shall be applied to the door support cable adjacent to the sponson work platform to simulate personnel stepping from the sponson work platform onto the cable, which eventually results in cable failure. The force and acceleration of such loading shall be derived from human factors information available. This loading shall be repeated for a total of eight impacts based on average estimated personnel required for maintenance and inspections.

3.1.3.6 Environmental Test - The lower personnel door shall be subjected to water and salt spray on the hinge and latch installation to simulate a high humidity and salt air environment, resulting in possible hardware corrosion. Sufficient spray shall be used to insure complete wetting of the hinge and latch installation surfaces.

3.1.3.7 Inspections and Functional Checks - Inspections and functional checks of the door, cables, support bracket, hinge, and latch installation shall be conducted at the completion of each test outlined above.

3.1.3.8 Test Sequence - Testing shall be conducted in programmed blocks designed to allow interaction of the simulated aircraft vibrations, impact loads, and open/close cycling while minimizing the number of test setup changes. A typical test block is presented in Section 4.2.

3.1.4 Main Rotor Pylon Work Platform Installation - The main rotor pylon work platform installation, P/N 65207-09004, has experienced the following problems in service:

(1) Delamination of the fiber glass surface.

(2) Failure of the latches to lock properly.

(3) Breaking and distorting of the hinges.

These problems have resulted from the combination of vibrations, normal usage, and abuse of the work platform.

One production work platform installation shall be subjected to the following laboratory tests, which are designed to duplicate problem modes experienced in the field. Prior to testing, the work platform shall be checked for proper operation of hinges and latches.

3.1.4.1 Steady and Vibratory Load Test - This section shall simulate in-flight load conditions. It consists of applying both a steady and a vibratory load, which results from aerodynamic loading and airframe vibrations. Measurements on an aircraft in-flight shall be made to determine the load magnitudes and frequencies. These loads shall be applied by eccentric mass driven by a varidrive motor with the work platform properly latched in the in-flight position. The expected mode of failure under these conditions is unlocking of the latches.

3.1.4.2 Cyclic Surface Loading Test - A cyclic loading test shall be used to simulate the loading conditions imposed upon the platform by a man moving

about on it. A load of 200 pounds shall be applied through a roller fitted with heels from Government-issue boots. This roller shall be cycled back and forth in the longitudinal direction upon the walking surface of the platform, which is fixed in the service position. This procedure is similar to that used to substantiate commercial aircraft floor panels. The expected mode of failure is delamination of the fiber glass, the principal problem with the work platform. Any deterioration of the fiber glass shall be properly recorded. Forty percent of the roller test will be run at an increased load of 400 pounds total, simulating two men working on the platform.

3.1.4.3 Cyclic Endurance Test - A cyclic endurance test in which the platform is manually cycled between the flight position and the service position shall also be included in the endurance test. The severity of the openings shall be regulated cognizant with normal usage. Abusive cycles shall be included with a free-fall opening and forceful closing. At the end of each test, the work platform will be checked for proper operation and any deterioration noted.

3.1.4.4 Environmental Test - Water and salt spray shall simulate a high humidity and salt air environment which results in possible hardware corrosion. The spray shall be applied periodically to insure constant wetting of the hinge and latch mechanisms.

3.1.4.5 Overall Interaction - The relative proportion of each section in the overall test shall be determined based on normal usage. Information concerning scheduled maintenance which involves the use of the pylon work platform requires that the platform be opened and closed twice for every 1½ hours of flight. Information concerning unscheduled maintenance is not available. Taking this into account, the platform shall undergo two open/close cycles per flight hour. The wheel shall be cycled four times for every open/close cycle.

3.2 Facility Requirements - Facility requirements for these tests include the following:

(1) Special jigwork for securing the respective assemblies during testing (References 2.6, 2.7, 2.8).

(2) A varidrive motor and eccentric mass for vibration testing.

(3) A hydraulic actuator and servo controller for vertical impacts on the personnel door (Reference 2.6).

(4) Pendulum fixtures for horizontal impacts on the personnel door (Reference 2.6).

(5) A test fixture and specially designed loading wheel for the work platform cyclic surface loading test (Reference 2.7).

3.3 Instrumentation Requirements - The following instrumentation shall be required:

(1) Strain gages positioned on assemblies for vibration testing. See Figure 2.

(2) Load monitor and automatic timing for vibration testing.

(3) Load cell for monitoring vertical impacts.

4.0 TEST LOAD SCHEDULES

4.1 Main Rotor Pylon Hinge and Cover Assembly -

(1) Functional check-out for hinged cover assembly

(2) 3.1.2.2 15 open/close cycles, 5 of which are abusive

(3) 3.1.2.1 4 hours of vibration testing: gage No. 2 ±400 psi @ 17 Hz

(4) 3.1.2.2 15 open/close cycles, 5 of which are abusive

(5) 3.1.2.1 4 hours of vibration testing: gage No. 2 ±400 psi @ 17 Hz

(6) 3.1.3.3 15 open/close cycles, 5 of which are abusive

(7) 3.1.2.1 4 hours of vibration testing: gage No. 2 ±400 psi @ 17 Hz

(8) 3.1.2.2 15 open/close cycles, 5 of which are abusive

(9) 3.1.2.1 8 hours of vibration testing: gage No. 2 ±400 psi @ 17 Hz

(10) Functional check-out of hinged cover assembly

NOTES: (1) Blocks shall continue in a similar manner until field failure modes are experienced.

(2) Normal cover maintenance shall be performed at the prescribed intervals.

(3) Water and salt spray shall be periodically applied to insure constant wetting of the hinge and latch mechanisms.

4.2 Lower Personnel Door

(1) Functional check-out of lower personnel door

(2) 3.1.3.6 Water and salt spray

(3) 3.1.3.1 8 open/close cycles

(4) 3.1.3.3 90 stair tread impacts

(5) 3.1.3.4 40 stair riser impacts

(6) 3.1.3.3 90 stair tread impacts

(7) 3.1.3.5 8 support cable impacts

(8) Functional check-out of lower personnel door

(9) 3.1.3.6 Water and salt spray

(10) 3.1.3.1 8 open/close cycles

(11) 3.1.3.3 90 stair tread impacts

(12) 3.1.3.4 40 stair riser impacts

(13) 3.1.3.3 90 stair tread impacts

(14) 3.1.3.5 8 support cable impacts

(15) Functional check-out of lower personnel door

(16) 3.1.3.6 Water and salt spray

(17) 8 hours vibration testing: gage No. 4 ±450 psi @ 17 Hz

(18) Functional check-out of lower personnel door

NOTES: (1) Blocks shall continue in a similar manner until field failure modes are experienced.

(2) Normal door maintenance shall be performed at the prescribed intervals.

4.3 Main Rotor Pylon Work Platform Installation

(1) Functional check-out of work platform installation

(2) 3.1.4.3 10 open/close cycles, 3 of which are abusive

(3) 3.1.4.2 80 cycles with the roller

(4) 3.1.4.1 4 hours of vibration testing: gage No. 3 ±500 psi @ 17 Hz

(5) 3.1.4.3 10 open/close cycles, 2 of which are abusive

(6) 3.1.4.1 4 hours of vibration testing: gage No. 3 ±500 psi @ 17 Hz

(7) 3.1.4.3 10 open/close cycles, 2 of which are abusive

(8) 3.1.4.1 4 hours of vibration testing: gage No. 3 ±500 psi @ 17 Hz

(9) 3.1.4.3 10 open/close cycles, 3 of which are abusive

(10) 3.1.4.2 80 cycles with the roller

(11) 3.1.4.1 8 hours of vibration testing: gage No. 3 ±500 psi @ 17 Hz

(12) Functional check-out of work platform installation

NOTES: (1) Blocks shall continue in a similar manner until field failure modes are experienced.

(2) Normal platform maintenance shall be performed at the prescribed intervals.

(3) Water and salt spray shall be periodically applied to insure constant wetting of the hinge and latch mechanisms.

5.0 TEST PLAN

5.1 Main Rotor Pylon Hinge and Cover Assembly

1. Install the hinge and cover assembly in the test fixture in accordance with Reference 2.8.

2. Verify proper operation of the cover assembly, and commence testing in accordance with the procedure outlined in Section 3.1.2 and the loading schedule in Section 4.1. Record completion of each test sub-block on data sheets supplied for the test.

3. Continue testing until all field failure modes are experienced or until 25 blocks have been completed. Record ANY failures!

4. Should breaking or failure of ANY cover assembly component be experienced, the provisions of Section 3.1.1 shall apply. Contact cognizant test engineer.

5.2 Lower Personnel Door Assembly

1. Install the lower personnel door assembly on the static H-53 aircraft in accordance with Reference 2.6.

2. Verify proper operation of the door assembly, and commence testing in accordance with the procedure outlined in Section 3.1.3. Follow the test load schedule presented in Section 4.2.

Record completion of each test sub-block on the data sheets provided.

3. Continue testing until all field failure modes are experienced or until 63 blocks have been completed. Record ANY failures experienced!

4. Should breaking or failure of ANY assembly component be experienced, the provisions of Section 3.1.1 shall apply. Contact cognizant test engineer.

5.3 Main Rotor Pylon Platform

1. Install the work platform assembly in the test fixture in accordance with Reference 2.7.

2. Verify proper installation and operation of the work platform assembly, and commence testing in accordance with the procedures of Section 3.1.4 and the loading schedule outlined in Section 4.3. Record completion of each sub-block on the special data sheets provided.

3. Continue testing until all field failure modes are experienced or until 25 blocks have been completed. Record ANY failures experienced!

4. Should breaking or failure of ANY work platform assembly component be experienced, the provisions of Section 3.1.1 shall apply. Contact cognizant test engineer.

6.0 QUALITY ASSURANCE PROVISIONS

6.1 Inspection - Both the test specimens and test installation shall be carefully inspected to insure conformity, alignment, assembly procedures, etc., which could affect the test data.

6.2 Calibration - All measurement systems used in these tests shall be calibrated and shall display a current calibration sticker. All load cells and strain gages shall be provided with electrical resistance calibration systems (R-cells), which will be checked prior to each test.

6.3 Witnesses - USAAMRDL and Navy Plant Representative Office shall be notified at least 10 days prior to the start of testing to enable witnessess to be present if required.

6.4 Responsibility

1. The test engineer has overall system responsibility.

2. All testing shall be conducted in accordance with the provisions of ESM-F1-2005, "Mandatory Requirements for the Conduct of Structural Fatigue Tests."

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